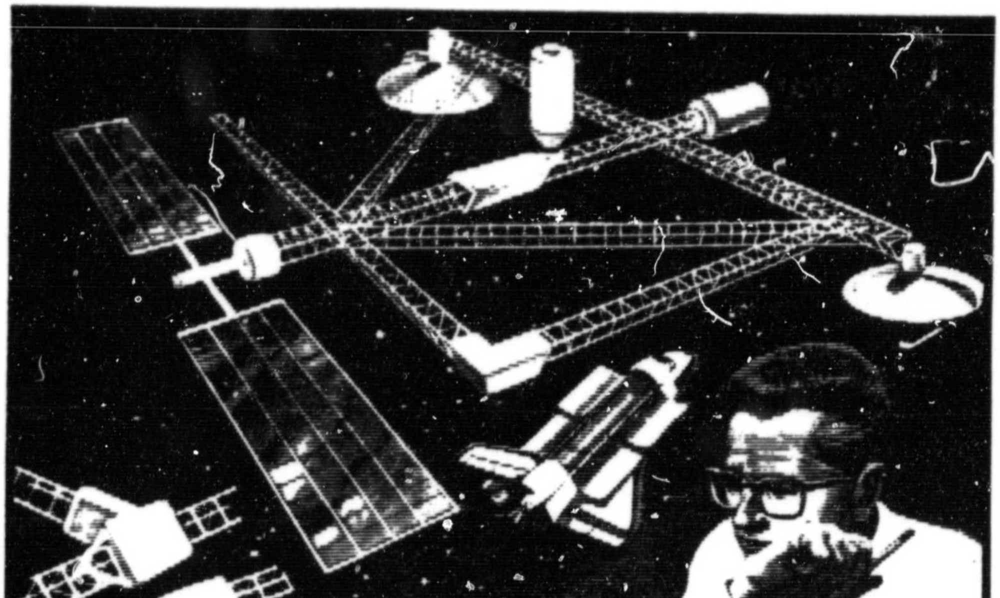
SSD 82-0121-1

DEVELOPMENT OF DEPLOYABLE STRUCTURES FOR LARGE SPACE PLATFORM SYSTEMS

Interim Report Volume I

NASA/MSFC CONTRACT NAS8-34677
AUGUST 1982



Space Operations/Integration & Satellite Systems Division
Rockwell International

SSD 82-0121-1

Development of Deployable Structures for Large Space Platforms

Interim Report
Volume I

Contract NAS 8-34677

August 1982

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FOREWORD

This volume and Volume II (SSD 82-0121-2) describe the study activities performed in support of the Part I study, "Development of Deployable Structures for Large Space Platform Systems" (NAS8-34677).

Volume II contains the preliminary design drawings that support the technical discussions provided herein.

This study contract was managed by Marshall Space Flight Center (MSFC) and was performed by the Space Operations/Integration and Satellite Systems Division of Rockwell International Corporation, located at Seal Beach, California. The study COR was Mr. Erich E. Engler. The study manager was Mr. H. Stanley Greenberg.

The duration of this Part I study was nine months; the start date was October 16, 1981.

The major contributors to this study were as follows:

- Design - R. Hart (Lead)
  - B. Mahr
  - A. Perry
  - J. Keech
- Stress Analysis - G. Lesieutre
- Thermal Analysis - T. Tysor
- Materials Analysis - R. Long
- Mass Properties - C. Griesinger
- Electric Power/Data Management - A. Gordon
- Electrical Utilities Integration - A. Le Fever
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- Technology Development - A. M. Pope

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INTRODUCTION

During the next decade, a revolution in spacecraft design will occur, resulting in large space platforms that will accommodate multiple payloads. Cost savings to users will occur through sharing of spacecraft utilities, ease of servicing, and the ability to change payloads. In addition, for geosynchronous communication payloads, platforms will reduce the crowding of this important orbital location.

The development of deployable platform systems is the most significant technology step in the direction of realizing these platform capabilities. Although the Shuttle allows payloads with much larger dimensions than other launch systems, the large dimensions of the platforms will, nevertheless, require extensive structural deployment to package it within the orbiter.

Much of the previous industry effort in large structures concentrated on orbital construction or erection. A recent Rockwell study, Space Construction System Analysis Study (NAS9-15718), developed the detail necessary to understand the difficulty of joining machine-made beams and integrating spacecraft utilities in orbit from the Shuttle. These studies pointed up the difficulty of erecting or constructing large platforms from the Shuttle. Consequently, ground integration of utilities into a deployable structure was selected by NASA as the first logical approach to platforms. NASA/MSFC has prepared a five-year plan to achieve technology readiness of deployable platform systems by FY 1986. Phase I of that plan is to identify the one or two most suitable deployable platform systems (Part I) and establish all the information necessary to plan and execute a follow-on-hardware development test program (Part II). On October 16, 1981 Rockwell initiated the study activities in support of Part I, with completion 9 months later - on July 16, 1982. Sections 1 through 4 of this interim report describe the pertinent study accomplishments for Part I.

Future missions such as the manned space platform require both pressurized and unpressurized volumes, respectively, for crew quarters and manned laboratories, and maintenance hangars. Deployable volume enclosures can minimize launch costs and enable use of volumes greater than those which can be transported by the Space Shuttle orbiter. On April 16, 1982, Rockwell initiated a 3-month add-on study of Deployable Volume Enclosures with the objective of identifying generic concepts for manned habitats, tunnels and OTV hangars. The accomplishments of that study are described in Section 5 of this interim report.

During the course of this conceptual study, 31 drawings were completed. The drawings are provided in a separate document, i.e., Volume II, SSD 82-0121-2, August, 1982.

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DESIGN APPROACH

The design approach employed in this study of deployable platform systems is developed to satisfy the objectives and guidelines as follows:

Objectives

- o Development and evaluation of generic deployable platform system concepts applicable to the focus mission LEO/GEO applications and foreseeable applications for the 1990 to 2000 time period

- o Establishment of a materials data base for structures and utilities systems compatible with LEO/GEO applications

- o Identification of the new technology development needs, schedules, and costs

- o Systematic/traceable selection of the one or two most suitable generic concepts

Guidelines

- o Automatic deployment - minimum EVA

- o Platform system - not just a structure

- o FY 1986 technology readiness - test-proven hardware

- o Generic - not a basepoint design

- o Versatility - can be used to build spacecraft of different configurations; "building block" approach (self-contained modules)

- o Distinction between LEO and GEO designs

- o Adaptable for a wide range of payloads

Of the above, the guideline of automatic deployment was the major design driver.

The entire concept development is directed toward automatic deployment of the entire platform system without use of a construction fixture or EVA. This approach is illustrated in Figure 1. The deployable platform system is comprised of several building blocks which are preassembled before flight to provide the defined spacecraft configuration. Each building block contains the deployable truss; integrated power, data, and fluid lines; a main housing; adapter; and deployment mechanization system. Attachments for payloads, reaction control system (RCS) modules and docking ports are provided on the main housings or adapters. The packaged platform system can be integrated into the orbiter with a pallet that serves as a control system module and contains batteries and telemetry, tracking, and command (TT&C) equipment. The system can be removed from the orbiter by the remote manipulator system (RMS) and placed on a handling and positioning aid (HAPA) or, depending on configuration, size, and shape, utilize both RMS and HAPA during the initial

deployment stages and during control system checkout. Subsequent to control system checkout, the platforms can be translated away from the orbiter (100 to 200 meters) for completion of the automatic deployment phase.

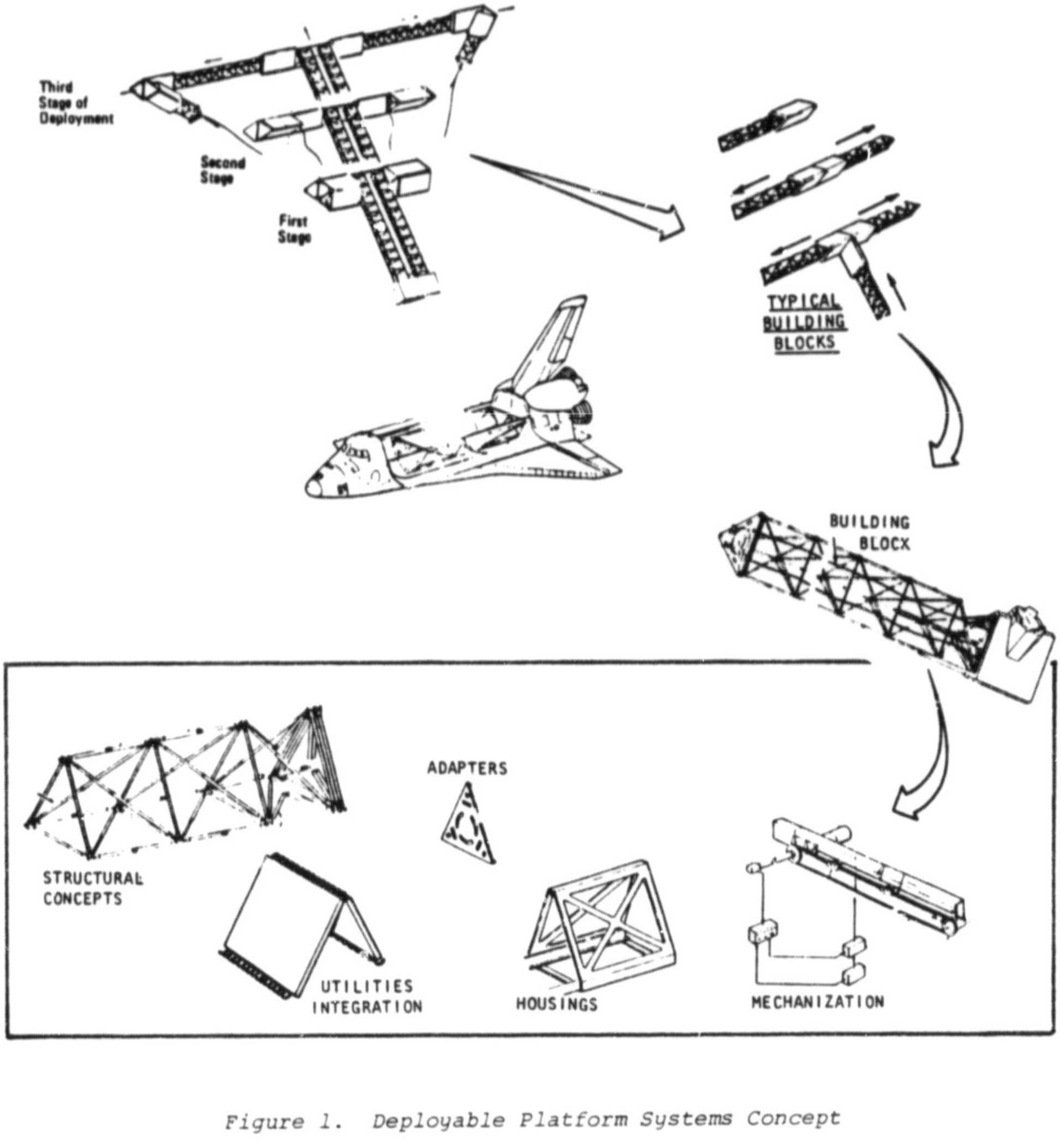

Figure 1. Deployable Platform Systems Concept

A major factor in the study approach is the requirements. Unquestionably, the regime of strength, stiffness, and utilities requirements is a major consideration. The requirements are extracted directly from the three focus missions (Figure 2, References 1, 2 and 3) and supplemented by Rockwell analysis and recognition of other potential applications.

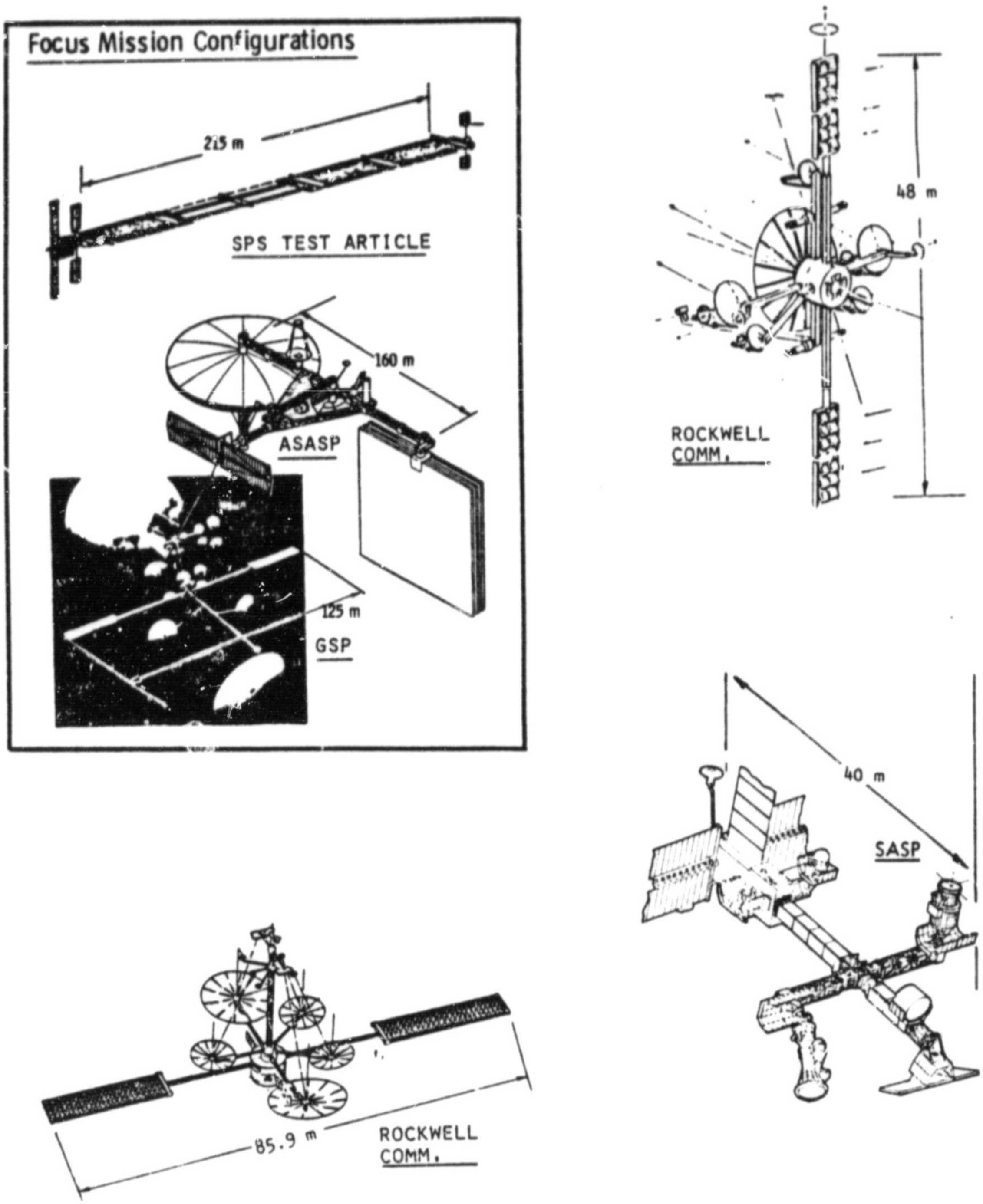

Figure 2. Focus and Miscellaneous Configurations

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The study approach was directed according to the study plan shown in Figure 3.

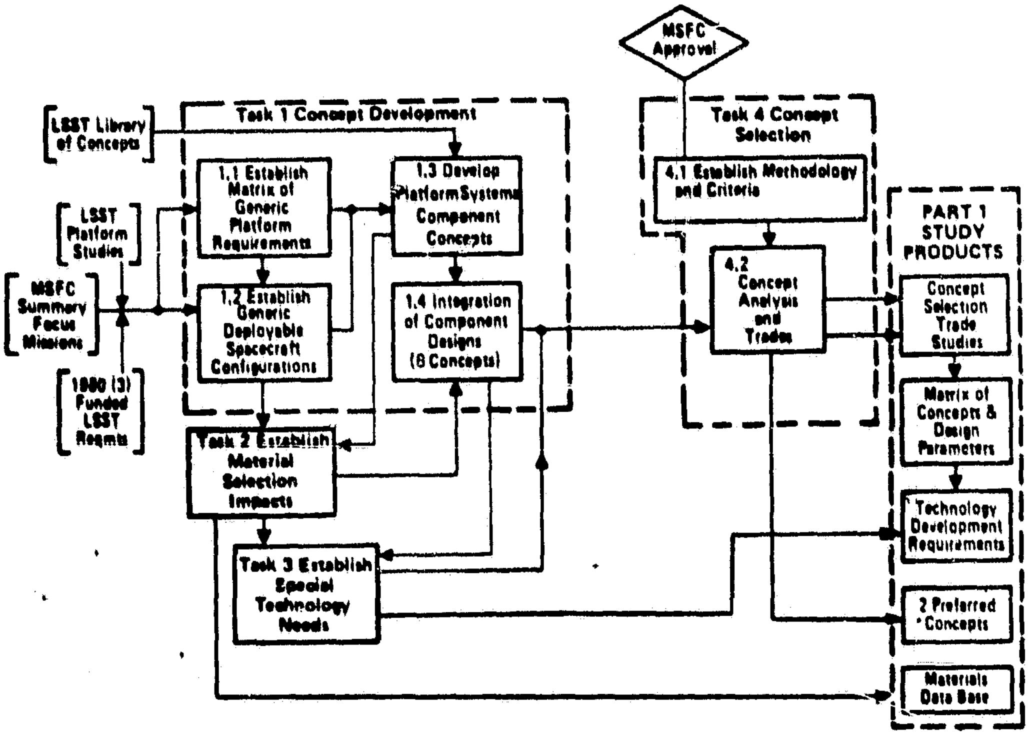

Figure 3. Study Logic—Part I

SUMMARY

This section briefly summarizes the major study accomplishments.

Figure 4 summarizes the deployable platform system accomplishments. In Subtask 1.1, the strength, stiffness, electrical, and fluid utilities requirements and additional requirements encompassing structural temperature limits, guidance and control, pointing accuracy, and propulsion were established. In Subtask 1.2, the generic configuration (to serve as a study tool) consisting of linear members was configured depicting the platform size, general arrangement, utilities distribution and docking ports. Also, investigation of an area platform constructed in such a manner that plate behavior is developed, resulted in termination of that concept for the reasons

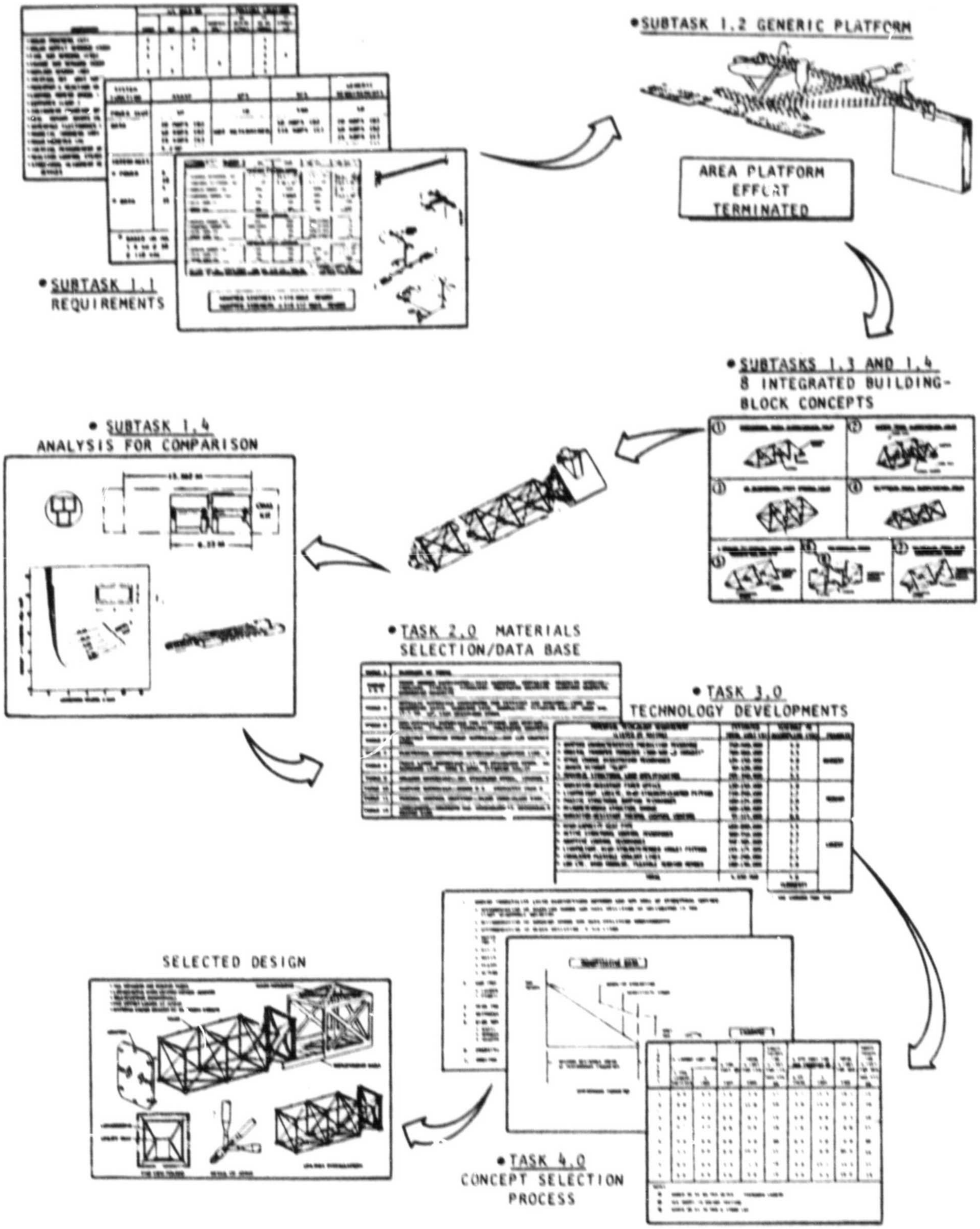

Figure 4. Summary of Deployable Platform System Accomplishments

delineated in Section 1.2. In Subtasks 1.3 and 1.4, eight candidate structure concepts with integration of utilities, and concepts for deployment mechanisms, housings to contain the mechanisms and structure, and adapters for payloads were developed.

All the concepts were integrated into eight building-block concepts. The eight building blocks were compared on the basis of packaging the generic platform into the orbiter, with supporting parametric structural, thermal, mass properties, meteoroid impact, and cost analyses. The results of these analyses are tempered by the consideration that the generic platform and requirements represent one design condition in the spectrum of platform applications. In fact, the generic platform and adopted strength, stiffness, and utilities requirement are at the upper end of the spectrum of foreseeable requirements for most platforms.

In Task 2, candidate materials most suitable for the deployable platform system components were identified, with establishment of a data base for these candidates. The data base is comprised of 10 tables of mechanical and physical properties.

In Task 3, new technology development needs were identified, prioritized, and scheduled, including the development cost estimates. Of the 16 new technology items identified, no show-stopper is apparent.

The foregoing data were used in the concept selection process of Task 4. The concept analyses compared the designs on the basis of design versatility, cost, thermal stability, meteoroid impact suitability, reliability, performance predictability, and orbiter integration suitability. This selection process methodology, in conjunction with judgmental evaluations, resulted in the selection of Concept 6A, (Figure 5). The major features of Concept 6A are summarized as follows:

- o Building-block approach for automatic deployment of platform systems
- o Square shaped truss - most suitable for inter-building-block attachments; mounting of payloads, docking ports, and propulsion modules; and provides redundancy for meteoroid impact.
- o Circular tubes for all truss members - minimum cost construction for graphite composite construction
- o Minimal complement of utilities mounted on longerons
- o Trays for mounting of large complement of utilities - ease of initial installation, repair, replacement during total ground fabrication period - minimum truss structural design constraints imposed by utilities integration.
- o Square-shaped housing with reciprocating deployment mechanism
- o Bay-by-bay deployment (to facilitate identification of deployment problem, if it occurs)

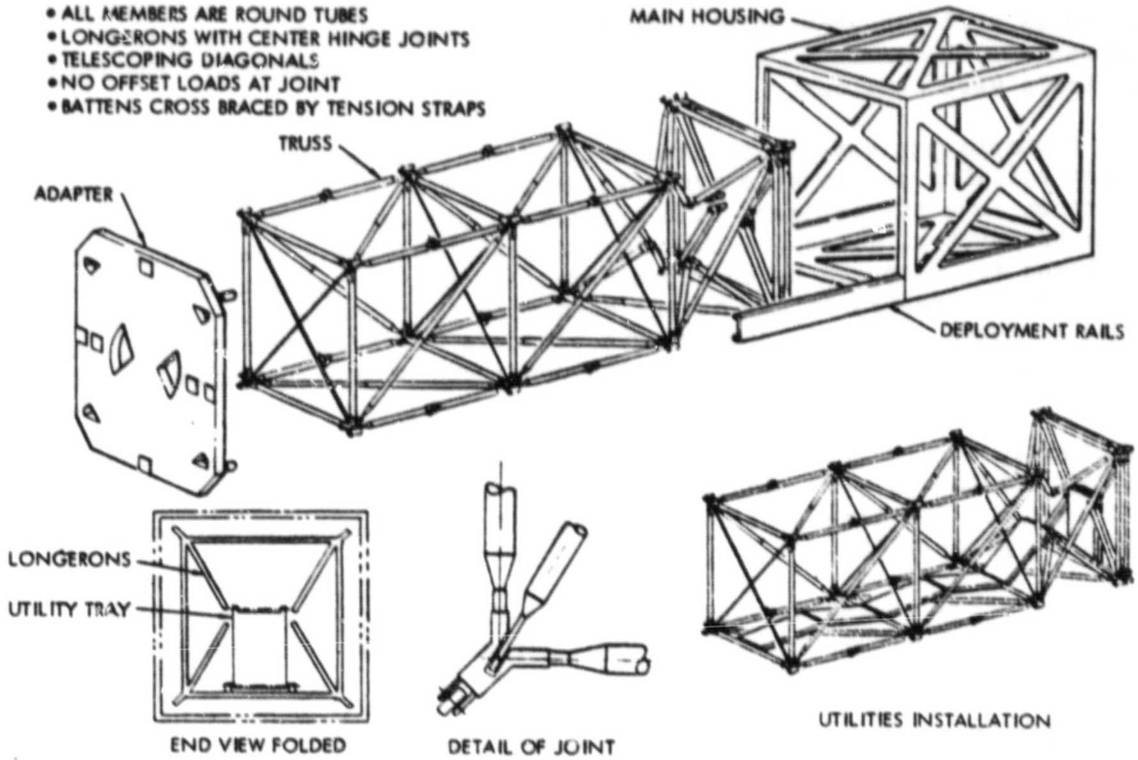

Figure 5. Square Truss with Modified Longerons (Concept 6A)

- Rail system for root strength during bay-by-bay deployment - permits orbiter berthing and orbiter vernier reaction control system (VRCS) firing (if necessary).
- Adapters for mounting of payloads with automatic electrical connector interface
- Payloads and propulsion modules attached using RMS

An example of a possible configuration achievable with this concept that is constructed of 1.75 and 2.75 m deep trusses is shown in Figure 6.

The major conclusions drawn in this study of deployable platforms systems are as follows:

- Deployable platform systems technology readiness for FY 1986 period is quite feasible (with appropriate funding)
- The building-block concept utilizing Concept 6A can effectively be used to construct LEO/GEO platforms
- Deployment is accomplished with orbiter RMS and/or HAPA without use of construction fixture
- NASA/MSFC goal of automatic deployment of platform system (not including payloads, RCS modules) is achievable

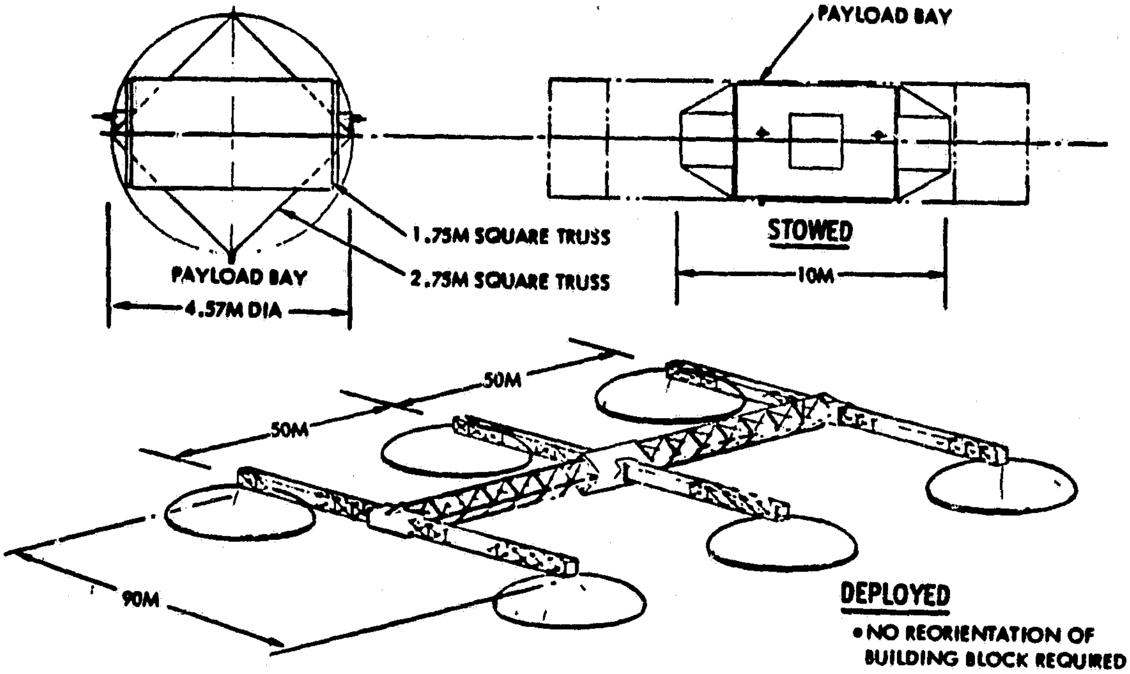

Figure 6. Possible Configuration (Concept 6A)

- o All candidate building-block concepts, applied to generic platform, are packageable into one orbiter and are well within 20,000 kg launch capability (28.5° inclination, 210 nmi)

- o The selected concept accommodates the upper regime of platforms size, utilities, and adopted strength requirements and is applicable to large range of reduced requirements

- o Accommodation of adopted stiffness requirements is dependent on the resolution of "Joint Slop" issue (most platform applications will have stiffness requirements well below adopted values)

- o No foreseeable unresolvable technology development requirements

- o Major extent of technology development for selected concept is applicable to alternate candidate concepts

Figure 7 summarizes the major accomplishments of the Add-on Deployable Volume Enclosures Study. The Space Station studies currently being performed at Rockwell are used for the establishment of requirements and applications for manned habitats and OTV hangars. Eight habitat and seven OTV hangar candidate configurations were developed. Three preferred configurations were identified for the manned habitats and developed in further detail (Section 5). Three

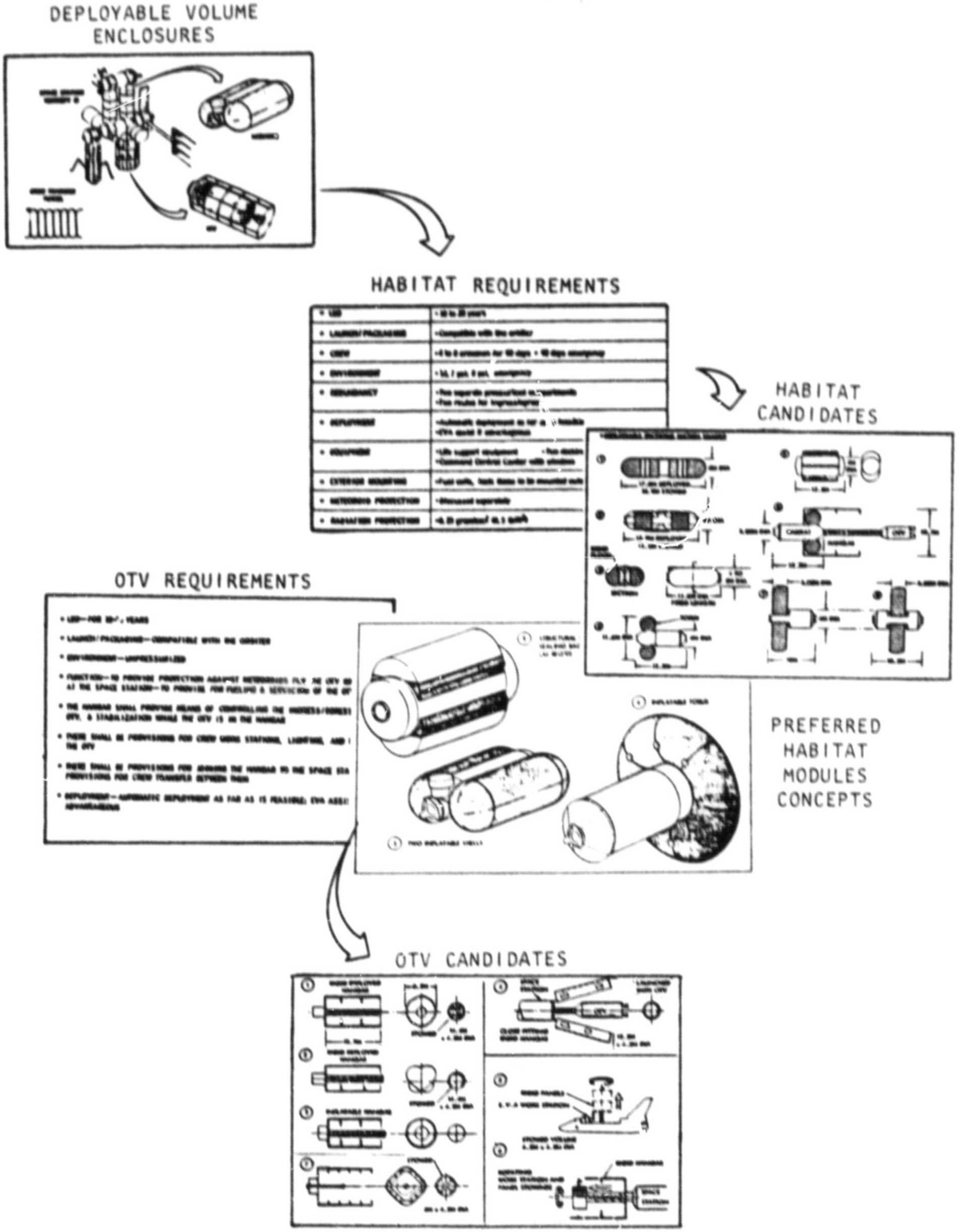

Figure 7. Summary of Deployable Volume Enclosures Accomplishments

OTV hangars were also developed in further detail. Further, the major design issues and new technology development requirements were identified.

The major conclusions drawn from this study are as follows:

- The application to habitats appears attractive - large useful volumes are achievable
- Metallic structures (can be sealed) can provide ample meteoroid, radiation protection, and equipment mounting surfaces, but are constrained by pressure loads/packaging requirements
- Inflatables alone are not sufficient - hard structure is required for mounting of consoles, orbiter and space station integration, and heat rejection
- Inflatables in conjunction with rigid core module provide a variety of feasible large volume designs provided

  Materials are suitable to crew safety/space environment; foam micrometeoroid stopping power is comparable to existing data; repair of punctures or use of meteoroid bumper is feasible; and adequate protection of the crew from radiation can be provided.
- Hangar requirements are ill-defined - OTV meteoroid protection alone is not sufficient justification
- Most attractive OTV hangar concepts appear to be metallic deployable/erectable or inflatable with foam core (provided stiffness is adequate)

1. CONCEPT DEVELOPMENT

This section describes the study concept development accomplishments which include establishment of platform system requirements (1.1), development of a generic spacecraft configuration (1.2), development of the building-block component concepts (1.3), and integration of these components into eight candidate building blocks (1.4). Section 1.4 also describes the application of the building blocks to the generic platform; packaging of the platform into the orbiter; and the comparative structural, thermal, mass properties, and cost analysis performed for the Concept Selection Trade Study (4).

1.1 DEPLOYABLE PLATFORM SYSTEMS REQUIREMENTS

This section describes the generic platform system requirements extracted either directly from the three focus mission studies (References 1, 2, 3 and 4) or determined by supplemental analysis from the data provided therein. In the course of performing the supplementary analysis pertaining to the ASASP and GSP requirements, the study managers of these studies were contacted for additional information and/or clarification.

The three focus mission studies provide four platform configurations as shown in Figure 1.1-1. Since Geostationary Space Platform (GSP) Alternative 1

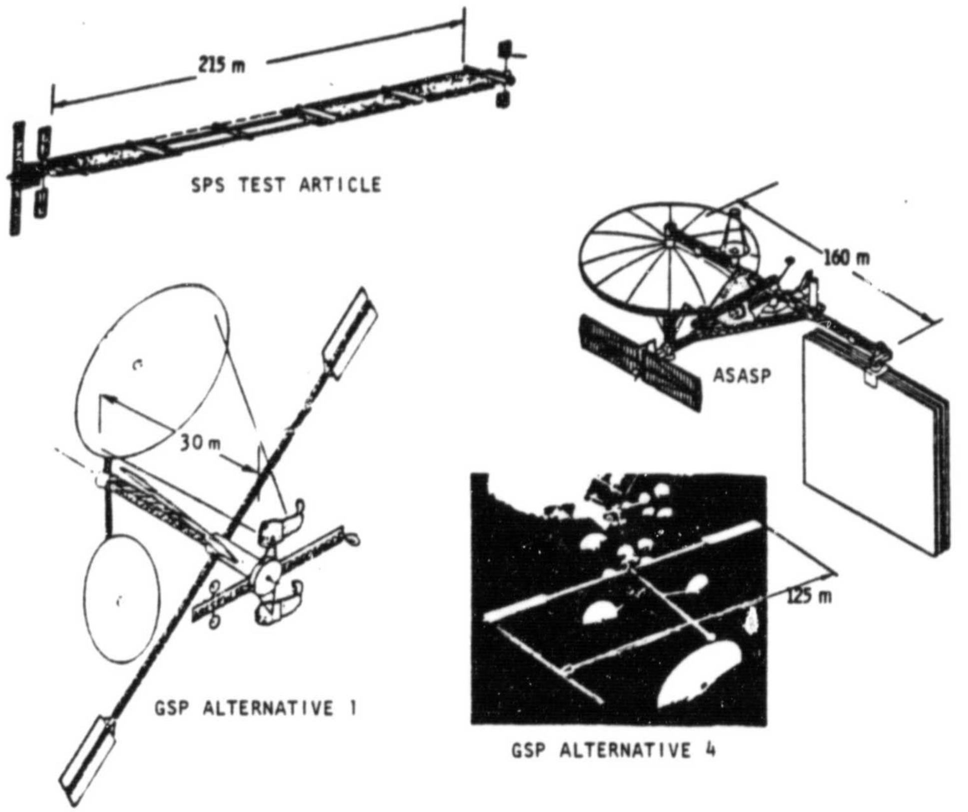

Figure 1.1-1. Four Configurations Extracted from Three Focus Mission Studies

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represents a configuration entirely constructable with either astromasts or supermasts, emphasis was placed upon the GSP Alternative 4 configuration which represents the high end of the spectrum in terms of size, strength and stiffness requirements. This configuration is comprised of three modules individually transferred to and joined together in GEO. Two of the largest of these three modules are shown in Figure 1.1-2. The information shown in Figure 1.1-2 and mass distribution data were extracted from the several books of Volume II, Reference 5.

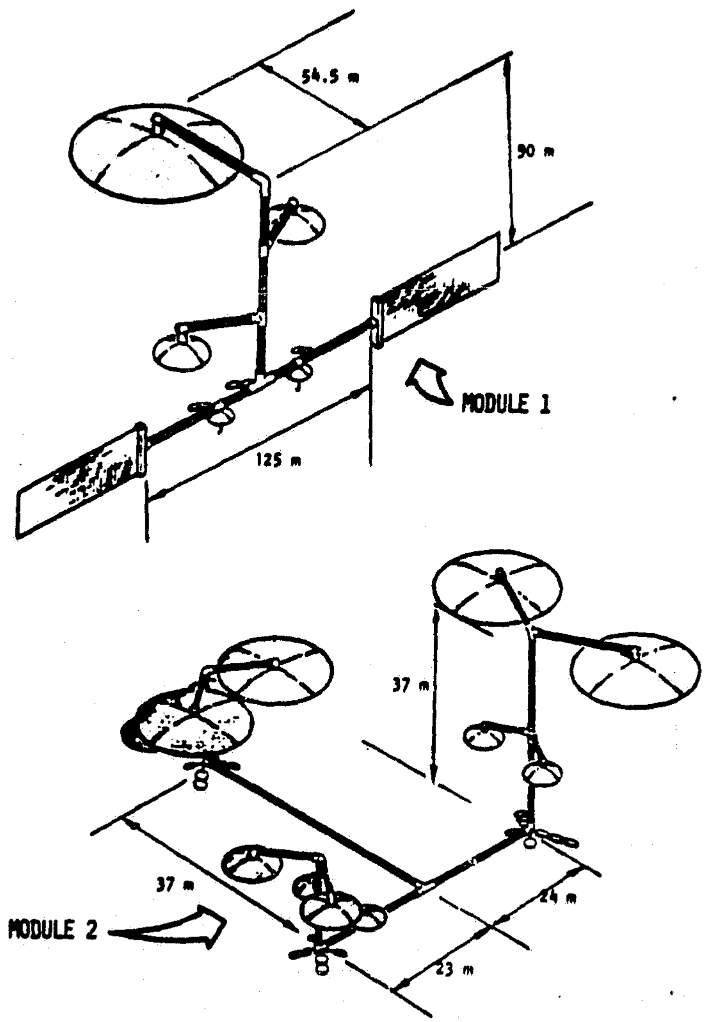

Figure 1.1-2. Modules 1 and 2 of GSP Alternative 4 Configuration

Additional to the supplementary analysis required to completely define the requirements for the missions, it was necessary to understand the derivation of the most severe requirements in order to assess the applicability of these requirements. The assessment was tempered by the consideration that this study is a technology development contract, with extension of the technology, as practicable, being a major goal. Further, many of the requirements stem from conditions where two equally possible alternatives to the design can exist and are dependent on the particular spacecraft configuration and systems trades. For example, thermal control of payloads can be accomplished with dedicated radiators at each payload or with the use of a central radiator and coolant lines integrated into the deployable structure. In such cases, in the the absence of any other information, the alternative requirement that advances the technology (in this case, integration of coolant lines into the structure) is included.

1.1.1 Strength and Stiffness Requirements

Table 1.1-1 summarizes the strength and stiffness requirements for the focus mission spacecraft configurations. The data shown were obtained by a combination of direct extraction from the focus mission documents and supplemental hand calculations. The data directly extracted are identified with a subscript "d".

Table 1.1-1. Focus Mission Limit Strength and Stiffness Requirements

PARAMETER \ PLATFORM	SPS	ASASP	GSP ALT. 1	GSP ALT. 4
DEPLOYABLE STRUCTURE MODULE				
• FLEXURAL STIFFNESS (Nm^2)	$17.3 \times 10^6{}_d$	√ $6.0 \times 10^8{}_d$	$2.8 \times 10^6{}_d$	√ $2.6 \times 10^8{}_d$
• TORSIONAL STIFFNESS (Nm^2)	$4.4 \times 10^4{}_d$	√ 1.1×10^7	8.2×10^4	√ $1.1 \times 10^7{}_d$
• BENDING MOMENT (Nm)	808_d	√ 9000	6570_d	√ $1.0 \times 10^5{}_d$
• TORSIONAL MOMENT (Nm)	18_d	√ 4900	3500	√ 1.8×10^4
• AXIAL LOAD (N)	200	500	√ 4660	3700
• SHEAR (N)	400	200	660	√ 5400
PAYLOAD INTERFACE				
• BENDING MOMENT (Nm)	1600	√ 4900	NEGLIGIBLE	90
• TORSIONAL MOMENT (Nm)	NEGLIGIBLE	√ 4900	NEGLIGIBLE	110
• AXIAL LOAD (N)	200	375	NEGLIGIBLE	90
• SHEAR LOAD (N)	100	500	NEGLIGIBLE	110
PROPULSION MODULE INTERFACE				
• BENDING MOMENT (Nm)	1200	1080	6190	√ 1.3×10^4
• AXIAL LOAD (N)	200	1780	1.05×10^4	√ 1.1×10^4
• SHEAR LOAD (N)	400	250	1050	√ 7850

1. Attachment of orbiter to platform accomplished by berthing.
2. Berthing loads are: bending moment - 1630_d (Nm), axial load - 1800_d (N).
3. Subscript "d" denotes data obtained directly from focus mission documents.
4. √ Denotes maximum values.

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The strength requirements are very sensitive to configuration and payloads size, shape, and mass distribution; and locations of propulsion modules for LEO stationkeeping or transfer to GEO.

The stiffness requirements are not only dependent on the payload and configuration size, shape and mass distribution but also the interplay between pointing system requirements, attitude control system (ACS) thruster levels, and control system design. For a basepoint design, trades between these systems can be made along a wide variation of parameters. Since this study is generic, such trades are not possible. However, the implications are recognized in the establishment of the requirements.

1.1.1.1 ASASP Flexural and Torsional Stiffness Requirements

The flexural stiffness value of 6×10^8 Nm^2 for the ASASP (Advanced Science and Applications Space Platform) configuration is extracted directly from Reference 4 as shown in Figure 1.1-3 and was determined to provide a minimum first modal frequency of 0.10 Hz. Since no information was available reqarding the required torsional stiffness, a hand calculation resulted in a value 1.1×10^7 Nm^2 for the same first modal requirement of 0.10 Hz. The total mass of the platform was 80,553 kg. The individual payloads, and mass moments of inertia were extracted from Reference 2.

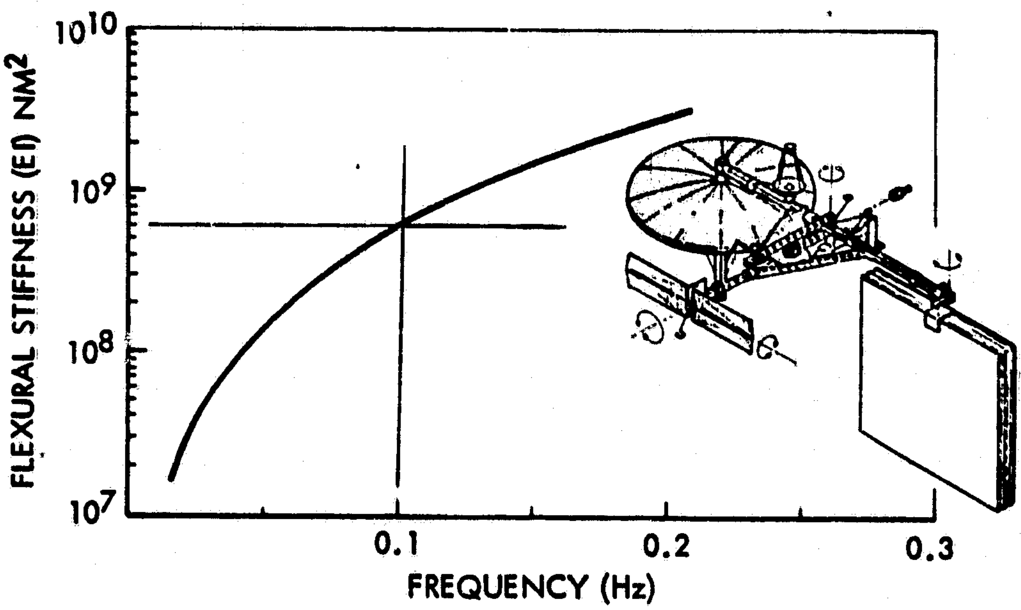

Figure 1.1-3. Flexural Stiffness Vs. Frequency (Hz) for ASASP

It is pertinent to note that during the orbiter packaging investigations, a NASTRAN modal analysis using an EI = 2×10^8 Nm^2, and GJ = 0.5×10^7 Nm^2, i.e., stiffnesses respectively 1/3 to 1/2 of the above quoted values, resulted in a first modal frequency of 0.043 Hz. Since flexural stiffness is the main driver, a frequency of 0.057 Hz would correspond exactly. However, within the broad context of this study, and the derived generic requirements, the model sufficiently confirms the validity of the stated requirements.

1.1.1.2 ASASP Strength Requirements

The ASASP strength requirements of 9000 Nm (bending) and 4900 Nm (torsion) occur during stationkeeping maneuvers. The propulsion module thrust of 890 N, directed as shown in Figure 1.1-4, induces the above specified loads. It is pertinent to note the relatively large torsional moment results from the large offset of the center of mass of the payloads.

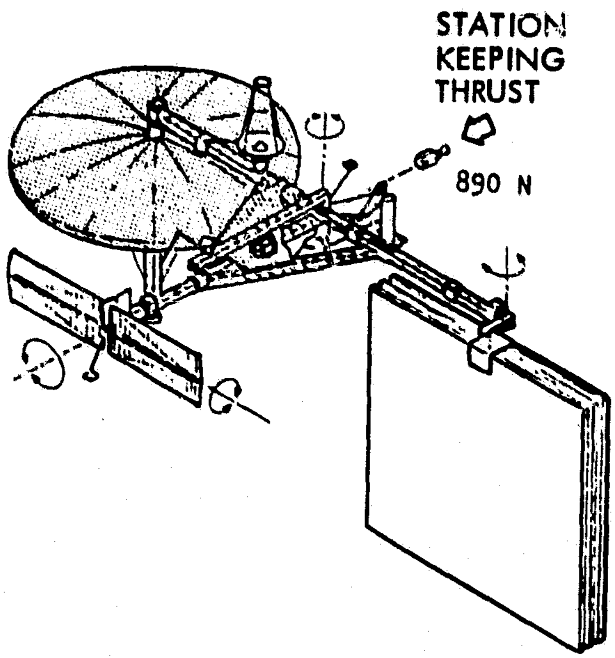

Figure 1.1-4. ASASP Configuration —Stationkeeping Thrust

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1.1.1.3 GSP Alternative 4 Flexural and Torsional Stiffness Requirements

The stiffness data shown in Table 1.1-1 are extracted directly from Table 3-6 of Reference 3. The requirements shown in Table 1.1-1 are the maximum requirements for members B and E (Figure 1.1-5). The requirements are based upon a restriction of the relative lateral displacement between the reflector and feed of 0.20 m due to RCS thruster induced loads. A hand calculation of the deflection due to inertial loads directed as shown compatible with an "approximate acceleration of .0003 g" (from Reference 3) indicated a deflection of 0.14 m (using amplification factor of 2). This proximity between 0.14 m and 0.20 m is quite adequate for this study. Of primary importance is the fact that the torsional induced portion of the deflection was more than 95% of the total deflection.

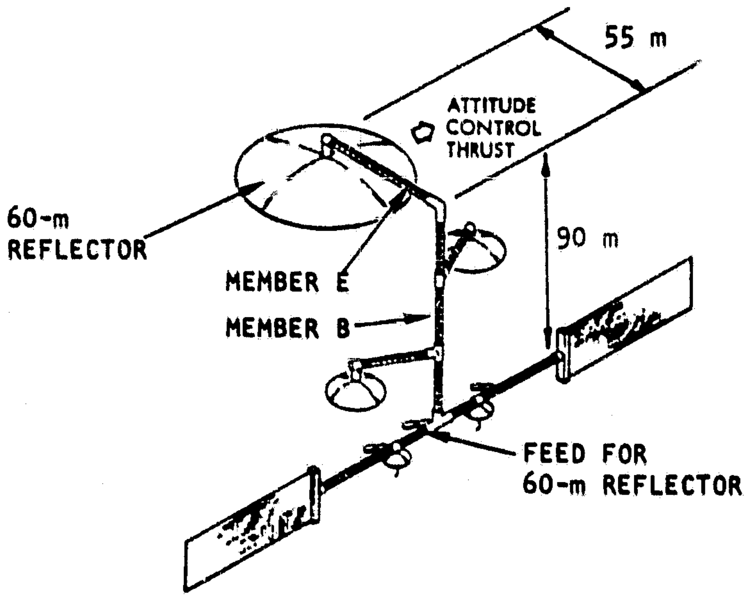

Figure 1.1-5. Module 1 (Alternative 4)
Attitude Control Thrust

Consideration was also devoted to the stiffness requirements to provide frequency separation with the control system. The following is an extraction from Reference 3 pertinent to this concern.

"A NASTRAN finite element model w generated for the Alternative No. 4 platform based on the individual module orbit transfer strength requirements. The model was comprised of 65 grid points, 64 structural elements, and 390 structural degrees of freedom. Natural modes and corresponding natural frequencies were determined for the system. The fundamental natural frequency of the system based on strength requirements is 0.019 Hz. A similar analysis of the Alternative No. 4 platform resized to comply with stiffness

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requirements would yield significantly higher natural frequencies. Again, caution must be exercised to ensure that the lower frequency vibration modes do not interact with the RCS and cause instability. As noted previously, control techniques can obviate this possibility."

1.1.1.4 GSP Alternative 4 Strength Requirements

The maximum bending moment extracted from Table 3-5, Reference 3 is 101,427 Nm for member H of Module 2 (Fig. 1.1-6). Torsional moments are not provided in Reference 3. The bending moment results from the GEO orbit transfer thrust of 6000 N directed as shown. The data presented in Reference 5 were used to construct a mass distribution model of Module 2 from which hand calculations confirmed the specified moment of 101,427 Nm, or more approximately stated, provided confidence in the mass distribution model used. A hand calculation of the torsional moment resulted in a value of 18,000 Nm. Both calculations include a factor of 2 for dynamic amplification.

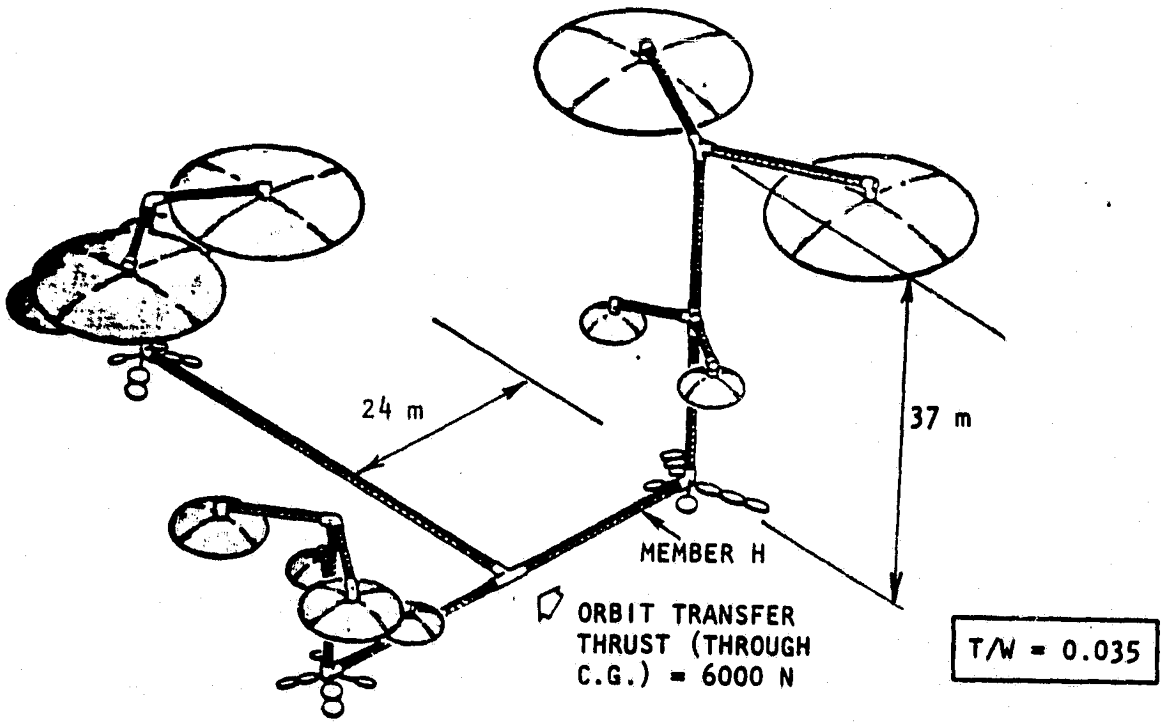

Figure 1.1-6. Module 2 (Alternative 4)
Attitude Control Thrust

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Table 1.1-2 illustrates the strength and stiffness requirements adopted for the concept development and structural analyses to be performed in this study. One set of requirements, i.e., the adopted requirements, is used to size the structure for the concept development drawings. However, the implication of order of magnitude variations of these requirements is studied in Section 4.

Table 1.1-2. Adopted Loads (Limit) and Stiffness Requirements

PARAMETER \ PLATFORM	SPS	ASASP	GSP ALT. 1	GSP ALT. 4
DEPLOYABLE STRUCTURE MODULE				
• FLEXURAL STIFFNESS (Nm^2)	$17.3 \times 10^6\ _d$	$\boxed{2.0 \times 10^8\ _d}$	$2.8 \times 10^6\ _d$	$\boxed{2.0 \times 10^8\ _d}$
• TORSIONAL STIFFNESS (Nm^2)	$4.4 \times 10^4\ _d$	$\boxed{.50 \times 10^7}$	8.2×10^6	$\boxed{0.5 \times 10^7\ _d}$
• BENDING MOMENT (Nm)	808_d	9000	6570_d	$\boxed{.25 \times 10^5\ _d}$
• TORSIONAL MOMENT (Nm)	18_d	7050	3500	$\boxed{1.0 \times 10^4}$
• AXIAL LOAD (N)	200	500	$\boxed{4660}$	3700
• SHEAR (N)	400	200	660	$\boxed{5400}$
PAYLOAD INTERFACE				
• BENDING MOMENT (Nm)	1600	$\boxed{4900}$	NEGLIGIBLE	90
• TORSIONAL MOMENT (Nm)	NEGLIGIBLE	$\boxed{4900}$	NEGLIGIBLE	110
• AXIAL LOAD (N)	200	$\boxed{375}$	NEGLIGIBLE	90
• SHEAR LOAD (N)	100	$\boxed{500}$	NEGLIGIBLE	110
PROPULSION MODULE INTERFACE				
• BENDING MOMENT (Nm)	1200	1080	6190	$\boxed{1.3 \times 10^4}$
• AXIAL LOAD (N)	200	1780	1.05×10^4	$\boxed{1.1 \times 10^4}$
• SHEAR LOAD (N)	400	250	1050	$\boxed{7850}$

$\boxed{}$ DENOTES ADOPTED STRENGTH AND STIFFNESS

The adopted stiffness requirements are based upon the following rationale:

- o The highest EI value listed for the ASASP of 6×10^8 Nm^2 is quite arbitrary, i.e., to achieve a first modal frequency of .10 Hz for a platform mass of 80,553 kg. It is unlikely a platform in excess of 40,000 kg will be required. Further, a first modal frequency of .03 Hz is still 100 times the LEO orbit disturbance frequencies.

- o Of the values (from GSP 4 Module 1) of EI = 2.6×10^8 and GJ = 1.1×10^7 Nm^2, only the GJ is the major requirement to limit the required deflection. That requirement is due to placement of the feeds for the 60 m reflector as shown in Fig. 1.1-5. Antennas with their own feed columns would preclude this requirement.

- o In view of the foregoing, and the technology advancement goals of developing designs up to the maximum practical/realistic requirements, the adopted stiffness values of 2×10^8 and $.5 \times 10^7$ Nm^2, respectively, for EI and GJ were selected.

. The adopted strength requirements were based on the following:

- The values of $1x10^5$ and $1.8x10^4$ Nm, respectively, for the bending and torsional moment obtained from the GSP alternative 4, Module 2, are regarded as unnecessarily high since the module is 20,600 kg (payloads to GEO by the year 2000 are not expected to exceed 6000 kg). Further, the loads can be reduced by reduction of the orbit transfer T/W (thrust to weight ratio), and/or minimization of dynamic amplifications. The latter two items are technology development needs, for antennas as well as platforms. The potential reduction of T/W from 0.035 to a possible 0.0137 is discussed as follows:

  Low-thrust liquid propellant systems are indicated for orbit tranfer applications for current and future missions to GEO as the need for maneuvering, start-stop operations and especially low thrust levels predominate as desirable characteristics.

  Low thrust and the resultant long burn times can mean larger gravity losses and increased propellant weight (Figure 1.1-7) for single burns. However, multiple perigee burns minimize gravity losses by reducing the burning arc and theta (Θ), the angle between the velocity vector and the local horizontal in the gravity loss term, $g_c t \sin\Theta$ (simplified).

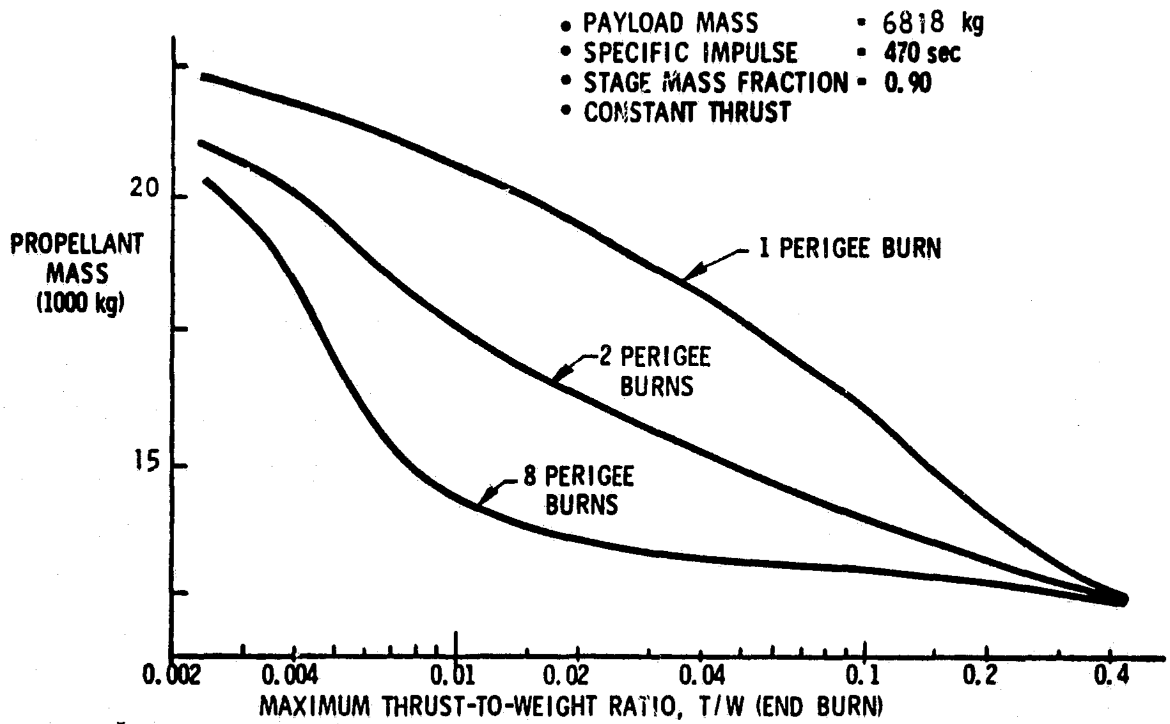

Figure 1.1-7. GEO Transfer T/W and Number of Perigee Burn Implications on Propellant Mass

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The use of multiple burns to minimize gravity loss (2% for eight burns versus 14% for single burn) is attractive in exchange for a somewhat longer transfer coast time (22 hours for 8 burns versus 6 hours for single burn). In many cases, this will be an acceptable compromise. The reduction in propellant requirements with multiple perigee burns at low T/W is dramatic, as shown for the payload indicated. Generally, propellant requirements increase as T/W decreases below the value of one, due to the aforementioned gravity loss effects. The propellant weight is decreased, however, as the number of perigee burns is increased above a nominal of one perigee burn. Eight perigee burns provide a substantial reduction in propellant requirement at low T/W; the maximum reduction occurring in the vicinity of T/W (final) from 0.01 to 0.02. These savings are sharply reduced, however, at T/W (final) values below 0.008.

o In view of the foregoing, and the technology advancement goals of developing designs up to the maximum practical and realistic requirements, the strength values of $.25\times10^5$ and 1×10^4 Nm, respectively, for bending and torsional moments were used.

In summary, the rationale for establishment of the adopted strength and stiffness requirements is presented above. In consideration of the overall range of requirements throughout the focus missions shown on Figure 1.1-1, the justifiable departure from the maximum values is minimal. Hence, it may be stated that the adopted requirements are at the high end of the total requirements spectrum.

1.1.2 Power and Data Utilities

Table 1.1-3 summarizes the adopted power and data utilities alongside of the corresponding data extracted from the focus mission studies. This requirement is used throughout the concept development.

Table 1.1-3. Generic Power and Data Utilities Requirements

SYSTEM FUNCTION	ASASP	GPS	SPS	ADOPTED	COMMENTS
POWER (KW)	50	10	490	50	SPS DEFINED AS SPECIAL CASE
DATA	20 MBPS (D) 50 KBPS (D) 25 KBPS (C) 4.2 MHz (A)	NOT DETERMINED	50 MBPS (D) 216 KBPS (C)	20 MBPS (D) 50 KBPS (D) 25 KBPS (C) 4.2 MHz (A)	(SCIENTIFIC) (HOUSEKEEPING) (TV)
INTERFACES					
POWER	6 NO. 0 28 NO. 2 4 NO. 14	6 NO. 3 4 NO. 13 20 NO. 18	396 NO. 10	6 NO. 0 16 NO. 2 4 NO. 14	
DATA	35 F.O.	34 NO. 18 TSP 58 NO. 26 TSP 144 F.O.	4 NO. 22 TSP	90 NO. 22 TSP 2 COAX 8 F.O. 100 F.O. (OPTION)	TO BE ACCOMMODATED BY DEPLOYABLE STRUCTURAL ELEMENTS

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The power lines are based upon a 50 kw total spacecraft power requirement and distribution system. The data utilities are based upon the scientific, housekeeping, and TV data needs as shown in Table 1.1-3 and discussed herein.

The power and signal distribution requirements are based upon analysis of possible payload combinations and altitudes. Table 1.1-4 presents a summary of the payloads considered. Included in the table are various physical characteristics and interfaces required to support the selected payloads. In a general sense, the distribution system is required to accommodate two classes of signals. The first is power at a relatively high level approaching 20 kW. The major power capability is distributed at 124-164 VDC, with lower loads at 30 VDC and 110 VAC, 400 Hz. The second class of signals are at fairly low levels and consist of command and data signals, digitally coded data transfer and low level closed-circuit TV (CCTV). The digital and video signal bandwidths are estimated to be 1 Mbps and 6.0 MHz, respectively.

Table 1.1-4. Reference Payload Group

PAYLOAD GROUP	MASS (KG)	DEPL. SIZE (M)	ALTITUDE (KM)	ORBIT (DEG)	POWER	DATA MGMT COMMAND (KBPS)	DATA MGMT DATA (KBPS)	NOTES
1. ATMOSPHERIC GRAVITY WAVE ANTENNA	3,000	100 (D)	>250	56-90	33 (kWh)	<25	10	SEE ①
2. PARTICLE BEAM INJECTION	3,000	100 x 100 (SQUARE)	400	56-90	3.3 (kWh)	<25	200 + 4.2 MHz (TV)	SEE ①
3. ASTROMETRIC TELESCOPE	4,500	2(D) x 18	400	28	1,000 (W)	<25	1,000	SEE ②
4. LARGE AMBIENT DISPLAY IR TELESCOPE	16,000	15(D) x 35	400-700	28-56	1,000 (W)	<25	7,000	SEE ③

NOTES
① EVA FOR MAINTENANCE AND DEPLOYMENT
② VIBRATION AND GAS PARTICLE SENSITIVE
③ VIBRATION AND GAS PARTICLE SENSITIVE + EVA

1.1.2.1 Power Utilities

The power utilities requirements are based upon the distribution system shown in Figure 1.1-8 and line lengths compatible with the generic platform (Section 1.2).

The wire size selection was conservatively based upon the worst case conditions at an operating temperature of 200°C, and line losses of approximately 5%. (Refinements in the analysis made at a later date to account for reduction of the operating temperatures to 20°C are discussed later.)

The 16 No. 2 requirement shown in Table 1.1-3 was derived from provision of 124 VDC at 290 A. For this condition, the primary path wire bundle was chosen to be eight No. 2 gauge (stranded) for power input plus eight No. 2 gauge (stranded) for returns.

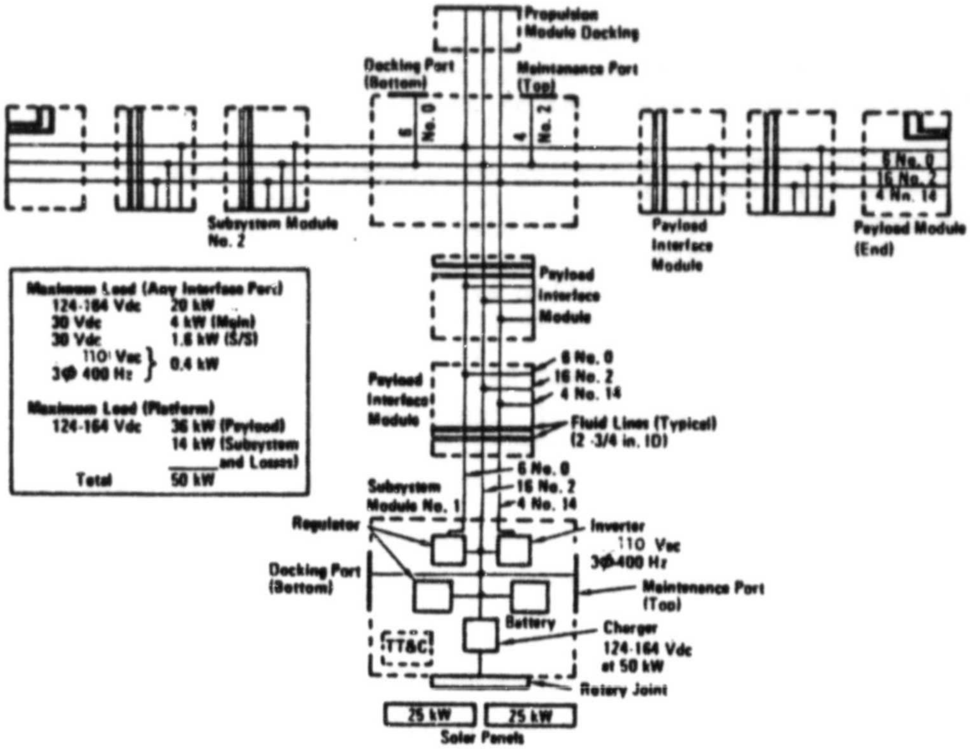

Figure 1.1-8. Platform Power Distribution Subsystem

Six No. 0 for the 30 VDC lines, and four No.14 for the 110 VAC, 400Hz lines were similarly determined. Later in the study, as part of an evaluation to determine the significance to the power utilities requirements of a 250 kW total spacecraft power requirement, refinements in the above analysis were made. The essence of this analysis was reduced electrical wire sizes due to replacement of the 30 VDC by 124-164 VDC, or replacement of both 30 and 124-164 VDC by 110 VAC, with converters and inverters at each port. For this reason and since the adopted requirement could be integrated into the designs, the adopted requirement was maintained. The investigation for a total spacecraft power load of 250 kW indicated essentially the same total circular mils as that of the adopted values provided the selected power lines operate at 460 VAC.

The basis for the foregoing conclusions is discussed in the following paragraphs.

The original evaluation assumed that the 30 VDC and the 110 VAC voltages were derived at a common location, located some distance from the solar array module. The effect of this assumption was to require that the deployable truss be capable of accommodating both the entire 50 kW at 124-164 VDC as well as accommodating 5.6 kW at 30 VDC and 0.4 kW at 110 VAC. In this original evaluation it was also assumed that the primary 124 VDC wire bundle temperature could rise to 200°C. Subsequent analysis indicated that a 20°C design limit could be maintained and, therefore, it was decided to redo the analysis of the basic concepts as well as all the other distribution concept options at the 20°C design point.

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The result of the revised basic analysis (summarized in Table 1.1-5) is to reduce the number of No. 2 AWG wires to eight. However, in the re-evaluation of the original analysis, the number of No. 0 AWG wires is increased to eight. Both numbers include their respective return wires or grounds. The total cross-sectional area of the wire bundle was reduced from 1,707,000 circular mils (in the original analysis) to 1,384,000 circular mils - a reduction of 18.4%.

An alternate to the basic wiring layout was also evaluated. In this case, the conversion to 30 VDC was accomplished within the solar power module with the resulting reduction in the power handling requirement of the 124-164 VDC wire bundle to 36 kW. At the same time, it was noted that the 30 VDC circuit may be required to accommodate 9.6 kW rather than 5.6 kW, so this increase in power level was also considered. The results of these changes are shown in the second column of Table 1.1-5. In this approach, the number of No. 2 AWG wires is reduced to six, reflecting the lowered power level, but the identified increase in 30 VDC power increases the number of No. 0 AWG wires to 12. The end result is a net increase in wire area of 21.5%. If the 5.6 kW requirements at 30 VDC are retained, a wire cross-sectional area of 9.5% is realized..

Four new options were considered during this study. Two of these options, Option A at 45.6 kW and Option C at 250 kW, assumed that all of the energy was provided at 124-164 VDC. The other two options considered the power is to be delivered at 460 VAC. Option B was rated at 50 kW while Option D provided 250 kW. The wire sizes for the ac power analyses were taken from a standard ac power handbook and adjusted for the voltage and current levels identified.

Table 1.1-5. Effects of Differing Power Load Assumptions Upon Wire Count

	OPTIONS					
	BASIC	BASIC (ALT)①	A②	B③	C④	D④
POWER LOAD REQMTS	50 kW @ 124-164 VDC	36 kW @ 124-164 VDC	45.6 kW @ 124-164 VDC	—	250 kW @ 124-164 VDC	—
	5.6 kW @ 30 VDC	5.6/9.6 kW @ 30 VDC	—	—	—	—
WIRE SIZE (AWG)	0.4 kW @ 110 VAC	0.4 kW @ 110 VAC	0.4 kW @ 110 VAC	50 kW @ 460 VAC	0.4 kW @ 110 VAC	250 kW @ 460 VAC
14	4	4	4	—	4	—
4	—	—	—	8④	—	38
2	8 (WAS 16)	6	8	—	58	
0	8 (WAS 6)	8/12	0	—		

① Electronics located in power module.
② Requires 124 to 30 VDC converters at each port, or responsibility of users.
③ Requires inverters & radiators (η = 90%) in power module, and rectifiers (η = 90%) at each port or within users' equipment. Losses of 10% at inverter will increase effective size of solar arrays (increase to 55 kW)
④ 5% line voltage drop permitted. (T = 20°C)

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The spacecraft configuration applied to all of the new options considered that the derivation of any voltage other than the primary distribution voltage, with the exception of the relatively low-powered ac in Options A and B, is to be accomplished at the user port or within the experiment system. Thus, in Options A and C the deployable truss must accommodate only two voltage levels, 124-164 VDC and 110 VAC. In Options B and D, only high voltage (460 V) ac must be accommodated. The advantage of eliminating the low-voltage distribution system is apparent, but it does require the addition of more equipment at each port.

The question of utilizing an ac distribution concept at a higher voltage was addressed in Option B. Again, it is possible to further reduce the total number of wires within a deployable truss at the expense of additional equipment, specifically rectifiers and transformers, utilized to provide the lower ac voltages and the various dc voltages. A further cost factor that should be appreciated, when using the ac concept, is the need to compensate for the additional losses introduced by the added equipment. With an estimated efficiency of 90% or less, the option will require a 10-15% increase in solar array to generate the necessary power.

Options C and D were evaluated to determine the impact of increasing the spacecraft experiment support power to 250 kW. Specifically, the dc distribution power level was increased to 250 kW at 124-164 VDC, while the ac system power level is specified at 250 kW at 460 VAC.

The effects of the higher power levels will, as expected, result in a larger number of wires. Total cross-sectional area of the wires varies from approximately 3,870,000 circular mils (in the case of Option C) to approximately 1,619,000 circular mils (Option D). In the case of the ac distribution system, it will again be necessary to appreciate the additional losses in the system caused by the port-located conversion equipment.

A final point of discussion is to examine the reasons why, if higher dc voltage will reduce loss effects, a higher dc primary distribution voltage (greater than 124-164 VDC) was not selected for the basic concept and for Options A and C. The major rationale for not selecting a higher voltage distribution system is the state of switching technology, particularly at the relatively high power levels specified. Switch devices capable of switching high dc voltages at high power levels are not yet available to the confidence levels needed to assure reliable operation. These switching devices include those simply used to control the power distribution as well as those used in dc-dc converters supplying the various lower voltages required. Several attempts at initiating technology programs to develop these equipments have been made with various degrees of success, but none have been completed. Accordingly, no hardware is available at the present time nor in the immediate future.

The second reason for not selecting a higher voltage is the impact upon the solar array. Solar arrays are comprised of many small solar cells interconnected in a series-parallel matrix to provide the needed power/voltage combination. Increased voltage levels increase the complexity of the solar array with the attendent reduction in reliability, life, and in poorer operating characteristics resulting from increased internal losses.

1.1.2.2 Data Utilities

The data management, control path, and the CCTV paths are essentially low power and are provided using three different forms of signal paths. Discrete signals or commands are routed utilizing 22 gauge, twisted, shielded pairs (TSP). Ninety pairs are included in the design of deployable structural elements; sixteen of which are preassigned as emergency control originating in the subsystem control module. The remaining 74 pairs are unassigned and may be used to provide interconnections between berthing stations.

The placement of these, or any signal wires, immediately adjacent to power cables is to be avoided. If the designs do not permit the separation of these cable groups, it is necessary to provide metallic separation (shields) to avoid electromagnetic interference caused by power switching.

The data management concept selected for this study presumes the use of an integrated, on-board data processing approach that uses a data bus design to minimize the overall number of discrete paths between satellite or platform communications interfaces. The proposed data bus link consists of four pairs of fiber optic cables. Each pair consists of an independent command and data channel. Four pairs are provided to accommodate reliability concerns (e.g., redundant paths in the event of channel failure) as well as providing for possible requirements calling for independent links to selected payloads. It is recommended that provisions to add up to 100 additional fiber optic cables to permit future system expansion be included in the element design.

The final requirement identified in prior studies is the need to provide at least two channels for routing of CCTV or high bit rate data to the satellite downlink communications system. The suggested coaxial is type RG-303/U.

In summary, it is pertinent to note that the data requirements delineated above are considered to be a generous complement of number and sizes. It is generally, however, consistent with the ASASP and GPS requirements. Further, spacecraft data needs during any program always press to the limit the ability of the structure to accommodate data needs. Hence, again, the generous complement of data utilities in the adopted requirements.

1.1.3 Fluid Utilities

The adopted fluid utilities requirement established for this study is two 2.0 cm coolant lines. Propellant lines are not a requirement for the reasons discussed subsequently.

1.1.3.1 Coolant Lines

Provision of fluid coolant lines is imposed as a requirement since location of radiators adjacent to a heat source may not always be practical. The requirement of two 2.0 cm lines was determined from the payload requirements of Table 1.1-4, which were extracted from Reference 2. For these payloads, the maximum power level was 25 kW.

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The use of a central radiator system would require that fluid lines be run along the structure between the heat source and the radiator and return. The pumping power required to circulate the coolant varies directly with length and power dissipation level and inversely with line diameter. For the reference mission payloads, a practical line diameter is 2 cm. Figure 1.1-9 presents the pump power to circulate Freon coolant through a 2 cm line over 40 meters and return for a range of power dissipation levels. As shown, a pumping power of less than 0.2 kW is required to circulate coolant to reject 25 kW of payload power.

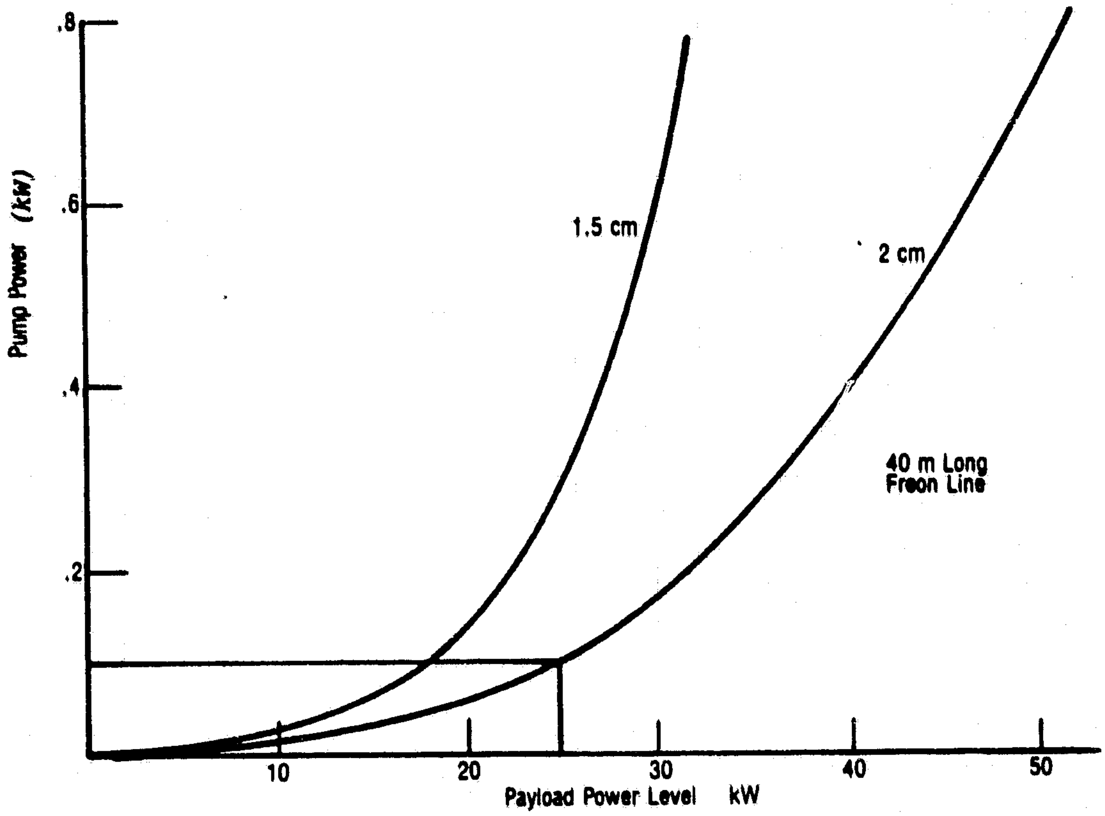

Figure 1.1-9. Pump Characteristics

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Insulation will be required for fluid lines used on the deployable platform to prevent freezing of the coolant during quiet periods. A quantitative estimate of the thermal control system (TCS) requirements for fluid lines is summarized in Table, 1.1-6 for a range of typical TCS coating techniques. These calculations assume the heat transfer area to be that of a 2-cm-diameter straight tube. The calculations show that a radiation barrier insulation (ε=0.04) is required to keep the drop in fluid temperature, due to line heat losses, at reasonable levels. A bare metal, such as aluminum, is unacceptable. These materials have high solar absorptance compared to its emittance. This would result in high tube surface temperatures that could cause localized boiling in the fluid line. A multi-layer insulation (MLI) blanket of about 5 to 6 layers would provide the proper emittance. The blanket could consist of concentric wraps of embossed metallized foil. A non-metallic outer layer may be desirable.

Table 1.1-6. TCS Requirements for 2-cm Fluid Lines

SURFACE COATING	EMIT-TANCE	LOSS OF TEMPERATURE (°C) FOR 40-m LINE			TIME TO FREEZE FLUID	HEAT LOSS AT FREEZ. TEMP.	TEMPERATURE (°C)	
		HEAT LOAD					SUN LOAD	
	ε	1 kW	5 kW	25 kW	HR	W/m	AVERAGE	PEAK
PAINT	0.9	52	10.4	2.0	0.5	3.3	24	122
COATING	0.5	28	5.6	1.2	1.3	2.0	24	122
BARE METAL	0.1	6	1.2	0.3	6.4	0.3	24	122
POLISHED METAL (α = 0.3)	0.04	2.4	0.5	0.1	13	0.15	219	381
MLI (α = 0.4)	0.04	2.4	0.5	0.1	13	0.15	-37	42

An alternate to the fluid coolant system is a heat pipe system. The heat pipe is a sealed heat transport device and does not require a pump to maintain its operation. A typical installation would use a 2 cm line to transport vapor and a 0.63 cm line to transport liquid. The technology of high capacity heat pipes is developing rapidly. Pipes with the capacity to transport heat loads of a few kilowatts over distances of a few meters are currently in development.

1.1.3.2 Propellant Lines

Integration of propellant lines into the deployable structure was not included as a requirement, although the use of distributed thrusters for either GEO orbit transfer or active modal control was considered.

The use of distributed thrusters has been suggested to reduce the bending loads imposed on the structure during orbit transfer. This advantage is not regarded as sufficient to offset the numerous disadvantages discussed below:

- Since the design must provide for failure of a thruster, each of the thrusters lines of force must pass through the platform center of mass for the engine out condition. For the sizes of GEO platforms

this would be applicable to possibly 2 to 4 thrusters. (Concepts utilizing more than 10 distributed thrusters as shown for solar power satellite (SPS) structures are not applicable.) Provisions for this condition would require excessive propellant.

- The additional cost of providing and installing the additional thrusters, and integrating the cross-feed propellant lines would be excessive. In particular, the folding and thermal control of propellant lines is a significant technology problem.

- The reduction of the bending and possibly torsional moments, while reducing the individual member loads, may not represent any significant weight reduction for stiffness-critical designs. The weight savings, if any, may be limited to the joints. The reduced loads on the joints and individual members could permit increased packaging efficiency. However, the loads can be reduced by stowage, which appears to be a simpler task.

It is pertinent to note the Large Spacecraft Systems/Propulsion Interaction Work Shop, held on October 22 and 23 in 1981, recommended the use of a single orbital transfer vehicle (OTV) with clustered thrusters for orbit transfer of GEO platforms. For the range of platform mass up to 6000 kg, a single OTV has the capability.

The use of distributed control thrusters is not considered appropriate for platform control in view of the following:

- The major spectrum of LEO/GEO mission requirements is achievable without special distributed actuators. Sufficient stability is attainable without distributed thrusters.

- For the special cases where unique payload precision pointing and high levels of stability are required special mounts such as initial pointing system (IPS) or annular suspension pointing system, gimbal system (AGS) will be provided.

If for some reason, distributed control is required, the preferred location for rotary or linear actuators is at the attachments between building blocks. Since the payloads are mounted only at the main housing or adapters, shaping of the basic truss is not required.

1.1.4 Control System

Control system requirements can be satisfied without the mounting of control system equipment directly onto the deployable truss. A control system module is provided to contain control system equipment other than that located at the payloads.

This review examined the three focus mission requirements with the basic platform functions and control philosophy as follows:

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- The platform provides a general-purpose base to support special-purpose (and multidisciplinary) payloads whose pointing and control requirements can be quite diverse.
- The platform provides gross stabilization and pointing control. Nominal attitude will be a local level and/or inertial orientation selected to minimize the platform disturbance torques and the resulting control system requirements.
- Payload precision pointing and high levels of stability will be provided by special pointing mounts (such as IPS and AGS).
- Specialized passive or active control for structural dynamic or figure control augmentation can be located at the housings or adapters (rather than on the deployable truss).
- The control system is designed to meet the most common recurring control problems - not rare or ill-defined special situations.

As evident from Table 1.1-7, it is feasible to locate all attitude control system equipment either in the control system module, the payload package, or at the building-block-to-building-block interfaces. Much of the equipment listed as mountable on the "control system module or payload package" can probably be mounted in the control system module for most NASA missions in which structural deformations are well within pointing requirements.

Table 1.1-7. Control System Equipment Requirements

	S/C USED ON				POSSIBLE LOCATIONS		
COMPONENT	ASASP	GSP	SPS	GENERIC ONLY	DEPLOY. STRUCTURE	CONTROL SYSTEM MODULE OR P/L PKG.	BUILDING-BLOCK INTERFACE
• SOLAR TRACKERS (ST)	X		X			X	
• SOLAR ASPECT SENSORS (SAS)	X	X	X			X	
• FINE SUN SENSORS (FSS)	X					X	X
• COARSE SUN SENSORS (CSS)				X		X	
• HORIZON SENSOR (HS)	X	X				X	
• INERTIAL REF. UNIT (GYROS)—(IRU)	X	X	X			X	
• MOMENTUM & REACTION WHEELS (MW)		X				X	
• CONTROL MOMENT GYROS (CMG)	X		X			X	
• COMPUTER (COMP.)	X	X	X			X	
• INSTRUMENT POINTING SYSTEM (IPS)	X					X	
• MISC. ROTARY JOINTS (RJ)	X	X	X				X
• INTERFACE ELECTRONICS UNIT (IEU)				X		X	
• MAGNETIC TORQUERS (MT)				X		X	
• MAGNETROMETER (M)				X		X	
• INERTIAL MEASUREMENT UNIT (IMU)				X		X	
• REACTION CONTROL SYSTEM (RCS)		X	X			X	X
• STRUCTURAL ALIGNMENT MEASUREMENT DEVICES				X		X	X
CONCLUSION: ALL ACS COMPONENTS ARE LOCATED ON ADAPTERS OR IN CONTROL SYSTEM MODULE.							

1.1.5 Structural Temperatures

Peak structural temperatures range from -100° to 80°C for LEO and -200° to 80°C for GEO. Figure 1.1-10 presents the peak temperatures calculated for the materials shown for an end of life ($\alpha/\varepsilon = 1$).

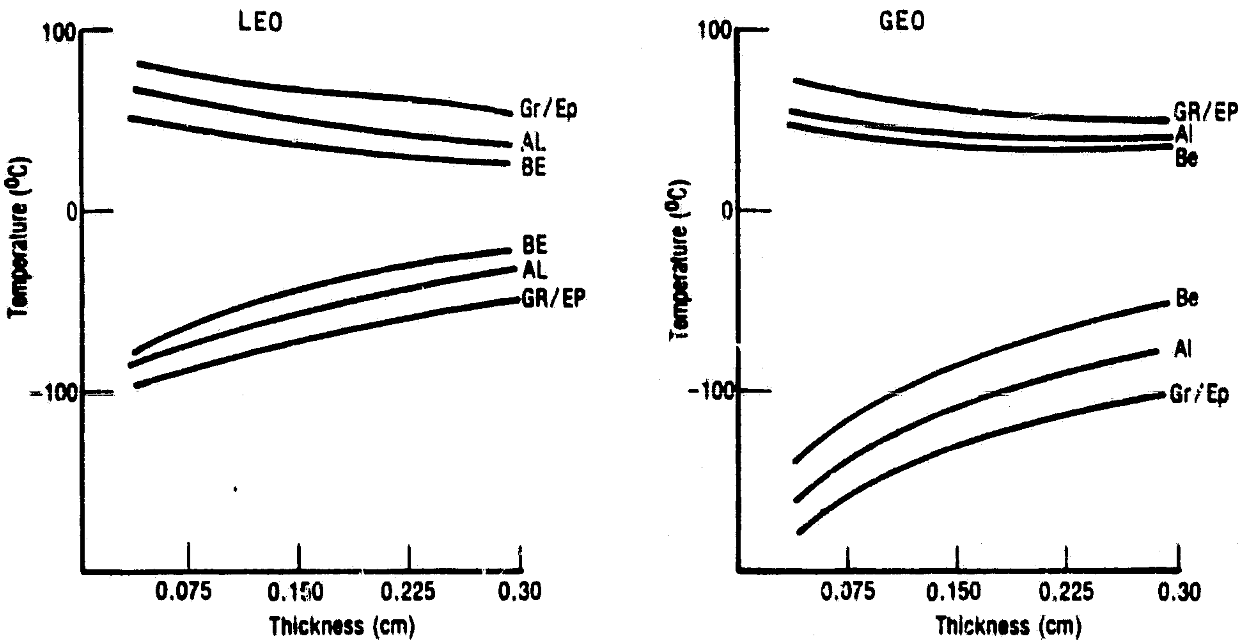

Figure 1.1-10. Maximum and Minimum Temperatures

The materials considered were aluminum, beryllium, magnesium, and several graphite and metal matrix composites. The peak maximum and minimum orbital temperatures depend upon the density times specific heat product for a particular material. For a given material thickness, the smallest density specific heat product will yield the largest maximum-to-minimum temperature range. Since a large number of materials is being considered, the materials were ranked according to this product. Beryllium had the largest value of this product, and graphite epoxy the smallest. Therefore, the properties of these materials were used in the analysis. Aluminum is used currently in spacecraft construction; therefore, data for aluminum are presented for comparison.

The member shapes considered were round, square, rectangular tubes and I-sections. The hottest structural temperatures occur when the member shape exposes its largest projected area toward the sun. A measure of this peak heating condition is the ratio of the solar projected area to the radiation area; therefore, the member shapes were ranked according to this ratio. For the member shapes considered, this ratio differs only slightly; therefore, only one or two shapes need to be analyzed. The square tube and the I-section

were selected for analysis. The data presented are for the square tube. The coldest structural temperatures occur during eclipse and are independent of member shape.

Individual truss members may be oriented at various angles with respect to the orbit plane and the sun. The orientation which exposes the largest solar projected area was selected for analysis.

Orbital temperatures were computed with a thermal math model (TMM). This model, for the square tube, consists of four nodes which are connected together by conduction and internal radiation. The nodes are connected to the external environment by radiation to space and by heat flow rates (QDOT's) for the incident solar, albedo, and earth emission fluxes.

1.1.6 Servicing

Servicing is assumed to be at one-year intervals, and consists mainly of replacing consumables or equipment with limited life. Changeout of payloads may be required from time to time because of "state of the art" or completion of mission.

The vehicles used for servicing are the orbiter for LEO and a teleoperator for GEO. It is expected that the teleoperator will use the same docking interface as the orbiter and will have a similar set of RMS.

The design of the platform, therefore, has to include docking provisions at strategic locations, near to payloads and service centers such as the control module. All items likely to be replaced must be designed for disconnect/removal/replacement using the RMS and/or EVA. Factors such as crew visibility, TV coverage, RMS reach, RCS plume effects and orbiter/platform interference need to be considered when designing for servicing.

1.1.7 Orbiter Integration

The orbiter is obviously an item of major concern in the design of a deployable platform. The orbiter is called upon to perform several functions:

- Transport the packaged platform to LEO
- Serve as a base for adding payloads, modules, etc.
- Checkout and troubleshooting the platform and systems
- Continued servicing and maintenance for a period of 10-20 years.

In performing these functions, the following factors are of importance:

- Docking/berthing clearances
- EVA capabilities and safety
- RCS plume effects
- RMS reach and capability
- TV & crew visual coverage
- Orbiter power available for platform
- Payload bay volume, c.g. and weight
- Mounting in the payload bay, cradles/pallets, trunnion and keel fitting location and loads
- Orientation during platform deployment, solar thermal, radiators
- Orientation, free drift, control authority
- Cost

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1.1.8 Environment

Materials for the deployable platform systems must have a minimum life of ten years in either LEO or GEO environments. Since transfer time to GEO is expected to be less than 24 hours with the use of chemical propulsion, the more severe LEO-to-GEO environment effects will be negligible.

Space environment effects are due to a combination of environments, some of which act at the surface and others which act throughout the volume (mass). Solar radiation, vacuum, and micrometeoroids are examples of environments which act primarily on exposed surfaces, while Van Allen Belt particles, solar flare particles, and the electron-produced Bremsstrahlung are examples of environments which act throughout the volumes of objects in space. Figure 1.1-11 illustrates the nuclear radiation components at GEO as a function of aluminum shielding thickness (the curves are similar for other materials) and Figure 1.1-12 illustrates the altitude dependence of the natural Van Allen Belts.

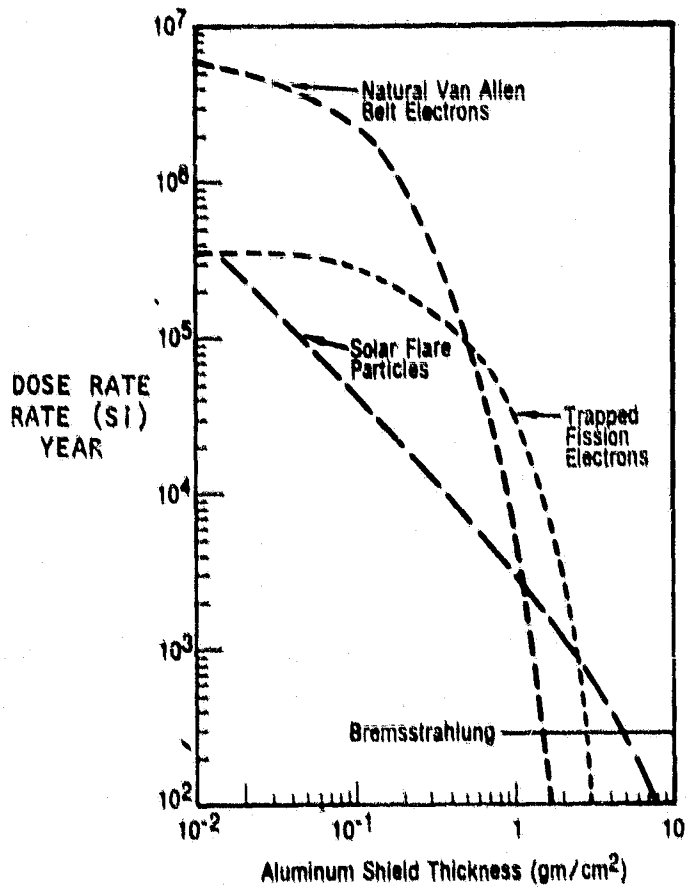

Figure 1.1-11. Radiation Dose Rates at GEO — Functions of Shield Thickness

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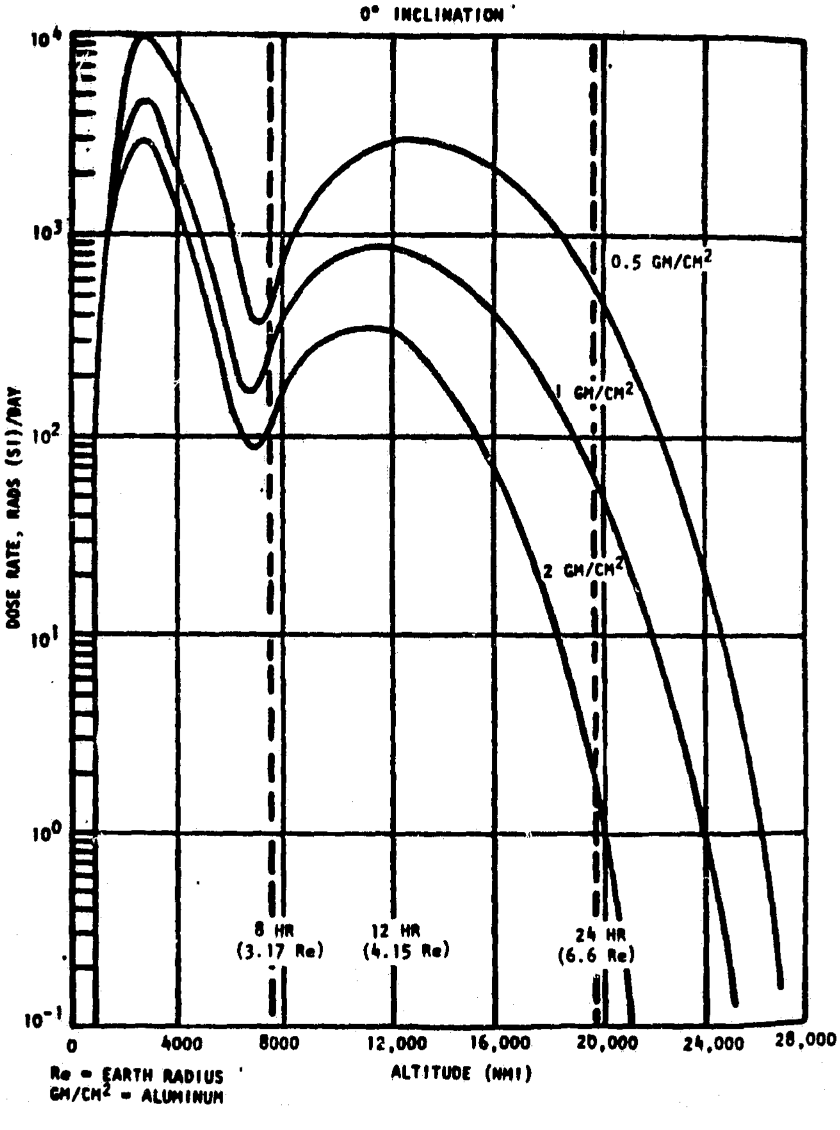

Figure 1.1-12. Natural Van Allen Belt Dose Rates, 0° Inclination Orbit

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The solar spectrum and vacuum are well known and are not included herein. Several materials can be severely impacted by both of these environmental properties, but thin protective coatings from other materials (such as thermal control coatings) can and do readily negate any excessive adverse effects.

Particulate or meteoroid flux can and does get significant as satellites increase in size and required service life. Figure 1.1-13 shows a time-averaged meteoroid flux at 1 AU from the sun. This meteoroid flux was obtained directly from Reference 6 and is sufficiently accurate for both LEO and GEO platform applications.

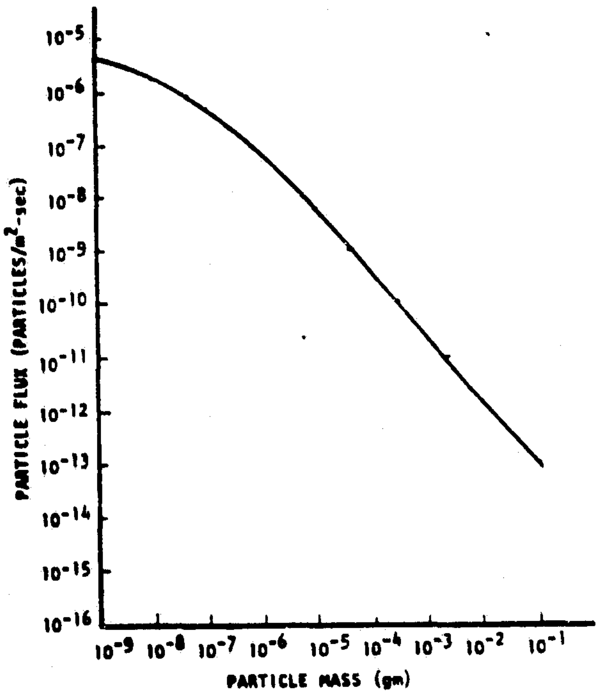

Figure 1.1-13. Natural Time Averaged Meteoroid Flux at 1 AU from the Sun

1.1.9 Payloads, Propulsion Modules, ACS Modules

Generally speaking, the linear platform is deployed without payloads, propulsion modules, ACS modules, or other large items which are not part of the basic platform. It is expected that they are added by the RMS to suitable interfaces subsequent to deployment of the platform structure. There are, however, possibilities of building some modules into the deployment system; depending on the size/shape of the module and the size/shape of the platform. If the circumstances are favorable, the module may be treated as another housing similar to a building block, and be incorporated onto the end of a

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truss. Some items may be mounted directly onto an existing control module or truss housing. If the module is to be added to the platform subsequent to platform deployment, an interface is provided complete with alignment features and all the structural/mechanical/electrical/fluid interconnects required for RMS berthing.

1.1.10 <u>System Pointing Accuracy</u>

The most stringent pointing requirement is delineated in Table 6 of Reference 4 for the GSP application. A value of 0.05 to 0.10° is listed.

Discussion with electronics specialists at Rockwell indicate pointing accuracies of 0.05 to 0.10° will be representative of most applications for the 1990 to 2000 time period, although a small number of applications will require accuracies of 0.03 to 0.02°.

1.1.11 <u>Requirements Perspective</u>

The design approach and requirements presented in Sections 1.1 through 1.10 are the basis for the generic platform development (Section 1.2); deployable truss, utilities folding/deployment, housing, and adapter concepts developed (Sections 1.3 and 1.4) and concept selection (Section 4). A perspective on these requirements are summarized below:

- The adopted strength and stiffness requirements are at the upper limits of the spectrum of requirements for the 1990 to 2000 time period.
- The generic platform represents the largest size of platform foreseeable for the 1990 to 2000 time period.
- The adopted complement of power utilities is representative of the maximum number and size of lines consistent with a 50 kW spacecraft (30 VDC and 124-164 VDC lines).
- The adopted complement of power utilities is also representative of a 250 kW spacecraft (460 VAC).
- The adopted complement of data utilities is quite extensive, but can vary considerably with payloads.
- During the platform design phase, the magnitude of data utilities is generally driven toward the maximum the structure can accommodate.
- The two 2-cm fluid lines requirement is representative of the maximum coolant fluid line diameter.
- A pointing accuracy of 0.05 to 0.10 degree is representative of a majority of systems for 1990 to 2000 time period.

This perspective is applicable to the focus missions (Figure 1.1-1) and to the spacecraft configurations shown in Figure 1.1-14.

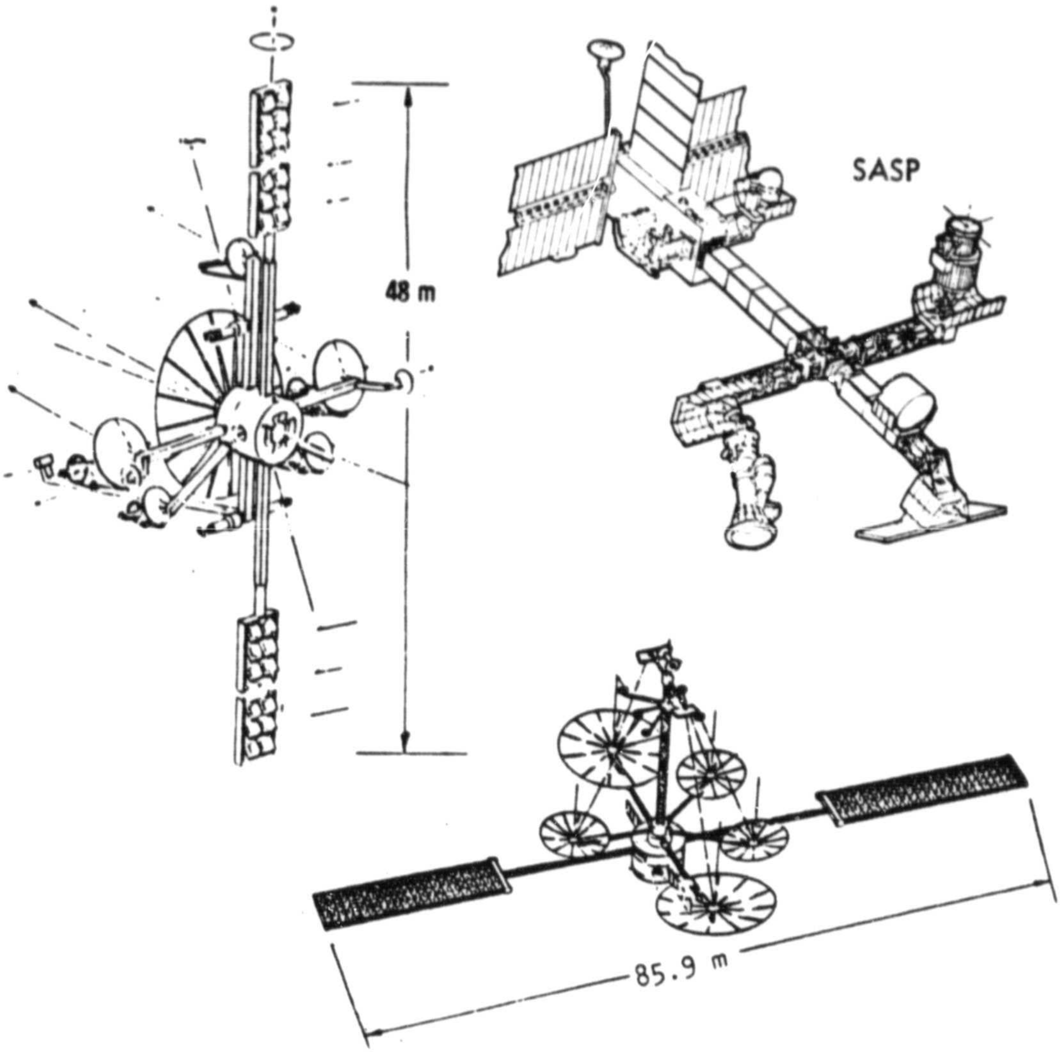

Figure 1.1-14. Rockwell Communications Configurations and SASP

1.2 GENERIC DEPLOYABLE SPACECRAFT CONFIGURATIONS

The generic linear platform (Figure 1.2-1) is based on the requirements established in this study. The purpose of developing such a platform is not to suggest that this is the best or only configuration available, but to display and understand the many problems encountered in designing a deployable system. Without such a configuration to study, there is a tendency to concentrate on the truss structure and overlook such important points as orbiter integration/packaging, deployment sequence and attachment of one section to another.

The configuration of the deployed platform is similar to the ASASP. The overall dimensions of the structure are 146 meters by 73 meters. The payloads shown (i.e., the reference payload system) are the atmospheric gravity-wave antenna, particle beam injection experiment, astrometric telescope, and IR telescope. Each of the four payloads is mounted at a "hard point" and not on the deployable truss itself. The design goal is to avoid mounting equipment on the deployable truss unless it is absolutely necessary. This policy has been followed throughout all the drawings. The electrical power for the reference payloads, and spacecraft systems, with allowances for line losses is 50 kW. The solar array is suitably sized and has the necessary rotary joints for two degrees of freedom. A radiator is shown mounted adjacent to the solar array module.

Although the reference payloads are LEO payloads, an orbit transfer vehicle is mounted on the aft end of the platform. Four reaction control system (RCS) modules are shown, of which two are mounted on short booms deployed from the control module, and two are mounted on structural hard points.

The control module (CM) serves three functions, i.e., it is a cradle or pallet for mounting equipment in the orbiter, a building platform for deploying the platform, and it is a part of the spacecraft platform and functions as the control center and houses equipment such as batteries, communications, data storage, and power conversion and control.

Provisions for docking/berthing of the orbiter are shown at the control module, at the solar array module, and at hard points on the structure, suitable for servicing payloads or propulsion units.

The structure consists of building blocks, arranged in such a fashion that they deploy sequentially or in unison to form the configuration shown. There is no "erection" or EVA involved, and there is no requirement for a building fixture or jig.

In the development of the several concepts for the trusses and building blocks, the generic platform was modified slightly to incorporate needs as they arose: for instance, sometimes there are two control modules and sometimes two trusses are joined into one building block.

Subsequent to the development of the generic linear platform, a generic area platform was studied in which the structural assembly behaves as a plate rather than as a number of beams connected together. The advantages expected to be gained of such a structure are:

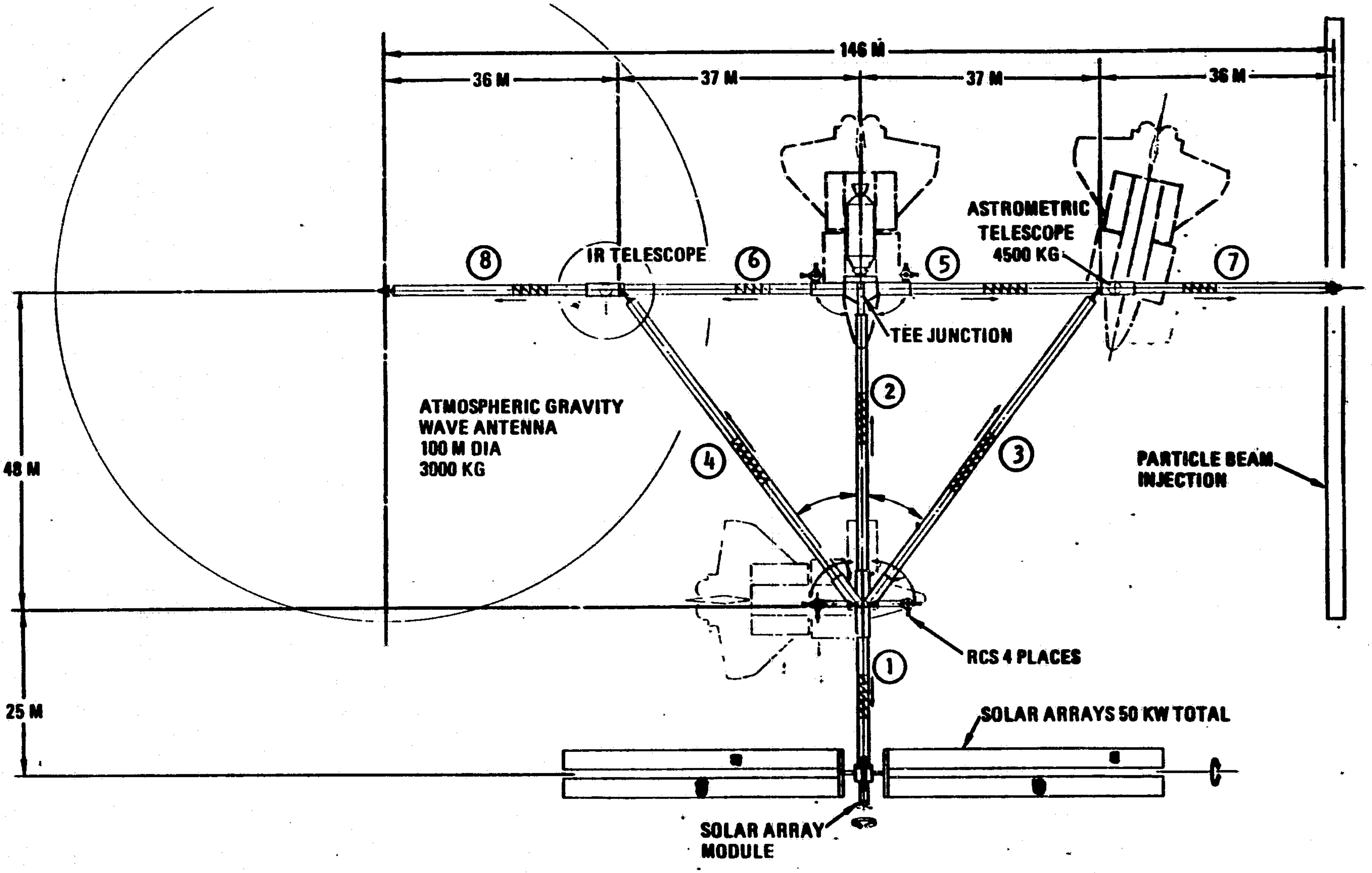

Figure 1.2-1. Generic Linear Platform

- A significant increase in stiffness provided payload and equipment attachments span to three node points.

- A significant decrease in torsional-type, thermal-induced deflections.

- Both of the above advantages may permit simplification of the control system, depending on the specifics of the design.

- Significant reduction in GEO orbit transfer-induced member loads, particularly for thrust in the plane of the platform. This is particularly favorable to joint designs.

- Significant increase in number of paths for routing of power and data lines.

The disadvantages of an area platform such as the Generic Area Platform Figure 1.2-2 are:

- A large area platform has drawbacks as compared with a linear platform when payload servicing/replacement is considered. The platform shown is about 1.5 meters deep. If the depth is increased, the accessibility problem is aggravated. This is a problem which is common to all large area platforms regardless of the type of structure or deployment method.

- Deployable area structures which deploy in two directions (i.e., length and width) possess certain drawbacks which are not present in other concepts:

  - Difficulty of controlling and holding the platform while it is deploying.

  - All bays deploy simultaneously in both directions, with root strength not achievable until full deployment.

Rockwell Inc. is not aware of the need for such an area platform application for the time period 1990-2000 other than solar arrays and antennas being developed in other study contracts. Therefore, with the concurrence of NASA/MSFC, the design of a deployable area platform was discontinued.

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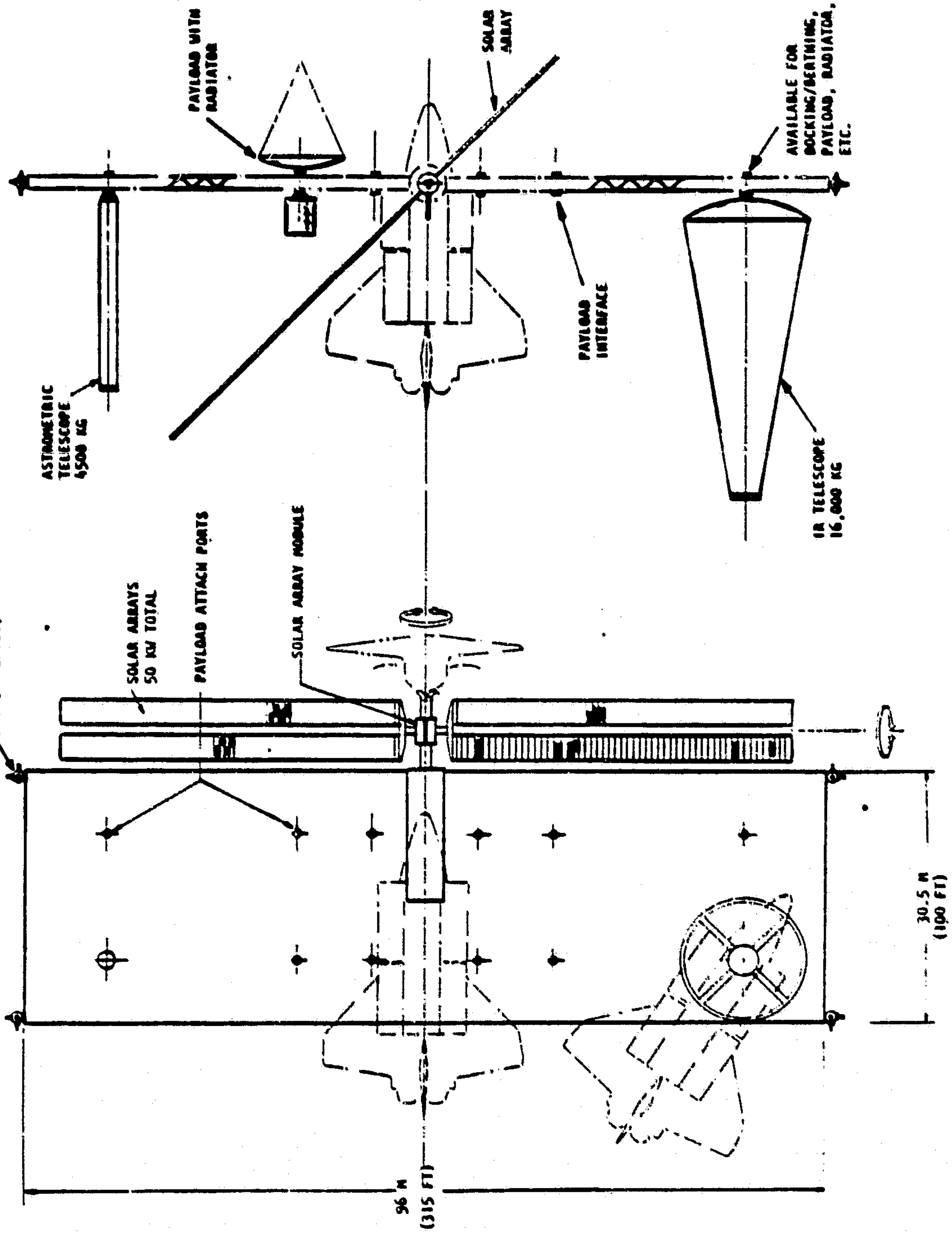

Figure 1.2-2. Generic Area Platform

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1.3 DEPLOYABLE PLATFORM SYSTEMS CONCEPTS

This section describes the individual component concepts developed that, in total, comprise the basic building block (Figure 1) from which automatically deployable platform systems can be constructed. The basic components (Figure 1.3-1) are the deployable truss, utilities integration system, deployment mechanization and rail system, the main housing into which the foregoing systems are folded during launch, and the end adapter.

The candidate designs for each of the aforementioned components are discussed in this section. In Section 1.4, the candidate components are integrated into total building blocks. Also, in Section 1.4, the integrated building-block designs were used to construct the generic spacecraft configuration, starting with packaging in the orbiter and ending with the fully deployed basic platform system.

The following section discusses the candidate designs developed for each of the building-block components discussed above.

1.3.1 Deployable Trusses

The establishment of the candidate deployable truss concepts gave serious consideration to the applicability of existing concepts. The search included review of in-house documents, the applicable documents listed in the four Large Space Systems Technology (LSST) bibliographies (Reference 12), and reviews of reports and discussions with the associated study managers of concepts recently developed/documented. Figure 1.3-2 illustrates most of the designs that were compiled as a result of that effort. Barring unseen proprietary designs, there is no deployable structure panacea, i.e., a structure that can be doubly folded into a very compact configuration (with utilities integration) that can be integrated into an automatically deployable platform system.

Figure 1.3-2 encompasses flight-proven designs, designs for which demonstration models have been made, and proposed concepts. These designs were reviewed on the basis of their suitability to satisfy the adopted strength and stiffness requirements and complement of utilities; compatibility with the total building-block approach; and compatibility with the single bay at a time deployment approach with maintenance of root strength during deployment. The designs using X-braced tension cables such as designs A, B, G, H, and K represent single-folded structures that are not compatible with the adopted GJ stiffness requirements.

The same comment is applicable to Concept F. A demonstration model of this concept was observed at the AIAA symposium in Long Beach, California, held on May 14, 1981. The design is a box truss containing circular longerons and I-section battens into which the longerons nest during stowage. The diagonal system uses X-braced tension straps preadjusted on the ground, so the preload is induced upon extension/locking of the longerons. Strap tension is approximately 45 N. No evidence of utilities integration was present in the model cell, which was approximately 4.6 m on its side. Demonstration of the model was presented by a series of staged slides. The basic deployment

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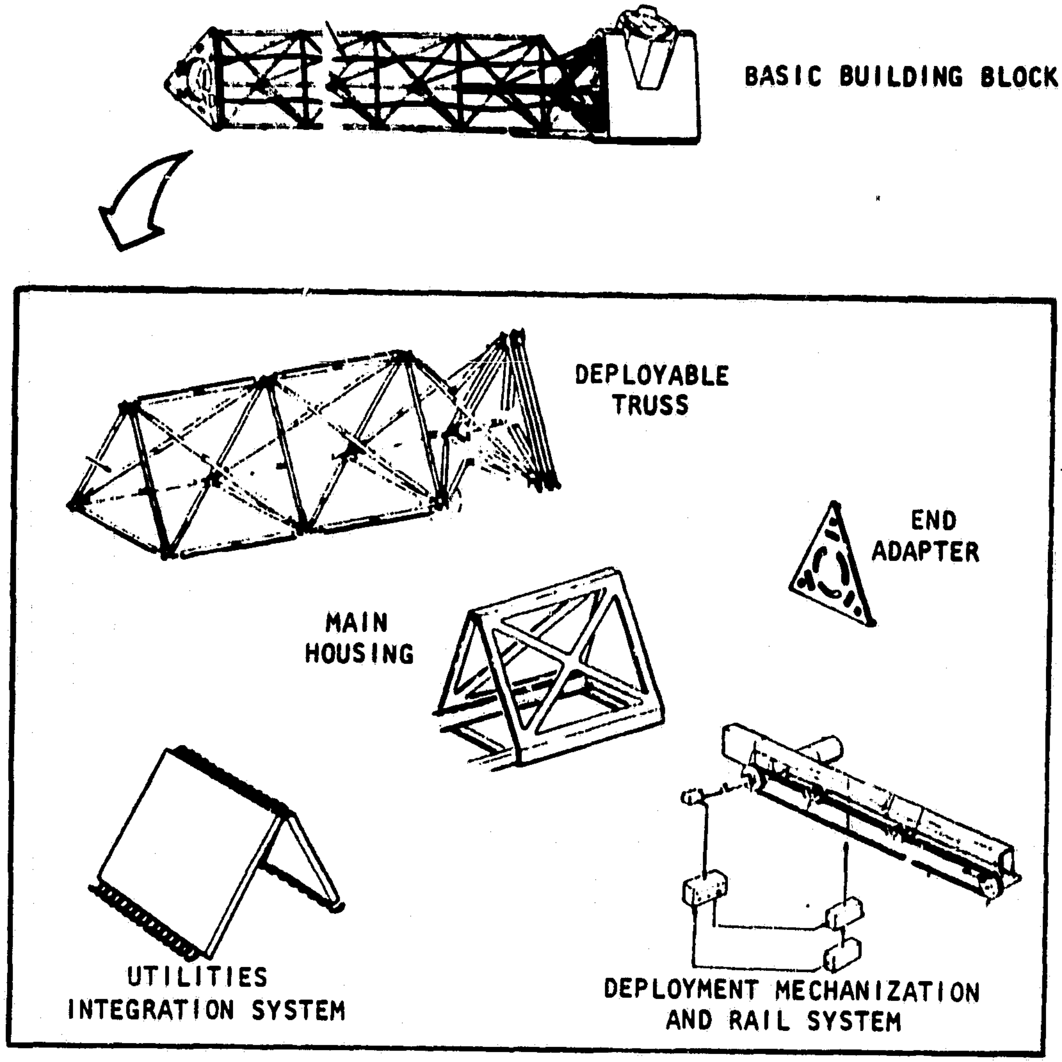


Figure 1.3-1. Deployable Platform System Components —Basic Building Block

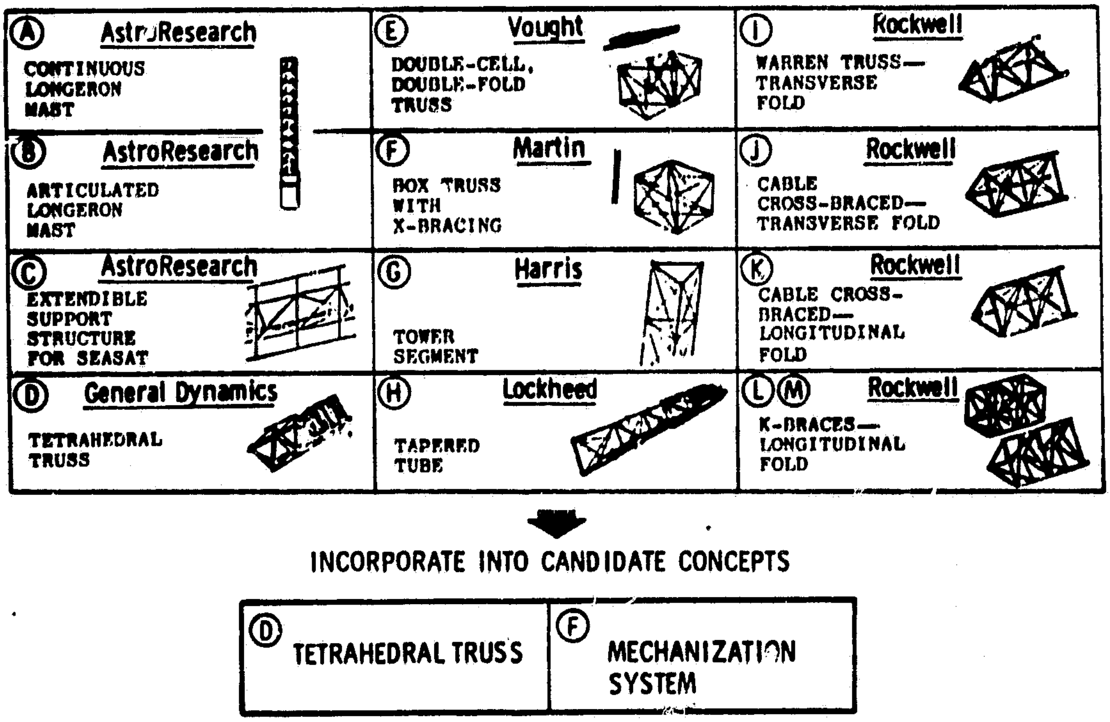


Figure 1.3-2. Review of Applicability of Industry Existing and Rockwell IR&D Designs

concept (obtained from Reference 8), however, for one bay at a time deployment utilizing stored strain energy (with retention of the remaining structure) was used initially in the development of Concepts 4 and 6.

Concept E has been successfully tested in the Neutral Buoyancy Tank at MSFC (References 9 and 11). This design has a limited length storable in the Shuttle cargo bay, since the longerons are not folded. This concept is regarded as essentially an erectable concept, and was not considered further, since no apparent method was foreseeable to use this concept in an automatically deployable platform system. The Rockwell-proposed designs (I and J) were not considered further for the same reason despite the attractive feature of non-folding longerons.

Concepts L and M were not pursued further because of poor packaging characteristics. Concept C (Reference 10) was not significantly different from Concept 1 (Figure 1.3-4).

Concept D represents a very efficient double-folded structure. The basic structure mechanization is quite simple (in the context of the complexity of double folding). Examination of the demonstration model (courtesy of General Dynamics) revealed well designed joints (concentric load paths and a minimum of material). This concept was therefore included (as Concept 3) in the candidate designs, although difficulty with integration into an automatically deployable spacecraft configuration was anticipated. Unquestionably, the design is extremely attractive in an erectable platform system that, through use of a fixture, joins together numerous deployable truss modules.

The development of new truss concepts resulted in the matrix of truss designs shown in Figure 1.3-3. This matrix of designs was derived to include the scope of single and double-folded designs, a design with X-braced tension cables that could satisfy the GJ stiffness requirements, and designs using triangular and square cross-sections. The matrix of these design variations is shown in Table 1.3-1.

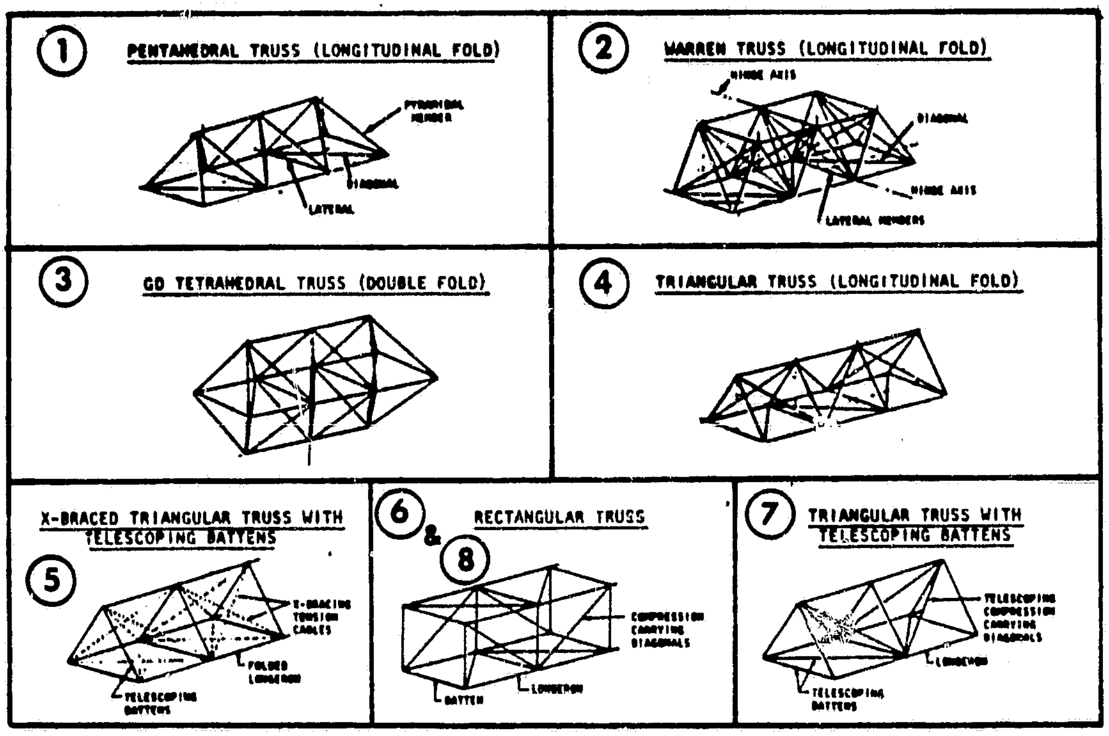
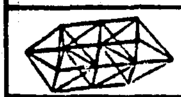
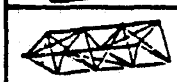


Figure 1.3-3. Candidate Deployable Platform Structure Concepts

All of the designs shown (Figure 1.3-3) satisfy the requirements in Section 1.1, with emphasis placed upon:

- o One bay at a time deployment to maintain root strength during all phases of deployment. A rail system is provided on the building-block main housing for root strength. Upon completion of deployment, the main load path from truss to truss is through the adapters and main housing (Section 1.3.4).
- o Satisfaction of the adopted strength and stiffness requirements in Section 1.1.1.
- o Capability to be integrated into the building block concept shown in Figure 1. For example, the matrix of design variations (Table 1.3-1) includes no designs that have only a lateral fold despite the significant advantage of such a design. Such a design could have clevis joints in the longerons approximately 17 m apart.

Table 1.3-1.
Matrix of Truss Variations

GEOMETRIC SHAPE	LONGITUDINAL FOLD COMPRESSION-TYPE MEMBERS	LONGITUDINAL AND LATERAL FOLD COMPRESSION-TYPE MEMBERS	LONGITUDINAL AND LATERAL FOLD TENSION CABLES
	1		
	2		
		3	
	4	7	5
	6 8		

The eight truss concepts shown in Figure 1.3-3 are discussed in detail in Section 1.4. The structural drawings are presented in Volume II. The main features in the development of each of these trusses are as follows:

- o Concept 1 (Figure 1.3-4) utilizes the kinematic advantage of the General Dynamics (GD) tetrahedral truss (for a single-folded design), the folding advantages of nesting the longerons and diagonals into the pyramidal members, and provision of clear space for utilities support trays.
- o Concept 2 (Figure 1.3-5) utilizes the high packaging efficiency of the offset longerons and shear panel as shown. The design requires only the diagonals shown in the top plane. Redundancy for meteoroid impact can be provided by addition of lower plane diagonals. The shear panel was used to provide lateral stiffness to the hinge line member to minimize deflection due to the offset longerons.
- o Concept 3 (Figure 1.3-6) is included for the reasons discussed previously.
- o Concept 4 (Figure 1.3-7) places emphasis on structural simplicity (in contrast to the designs of Concepts 2, 3, 5, 7, and 8), both in regard to member shape and kinematics. All the structural members are circular tubes, with folding of the longerons and telescoping of the diagonals as shown.

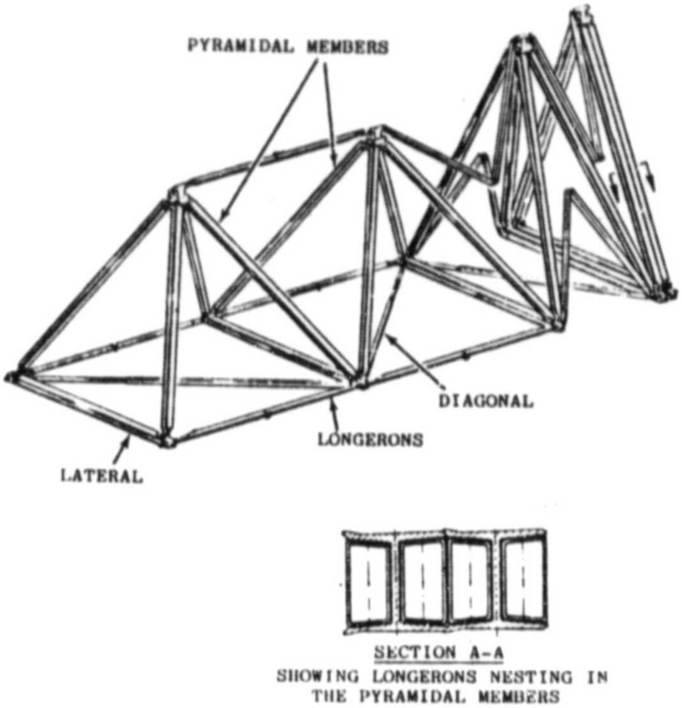


Figure 1.3-4. Concept 1—Pentahedral Truss

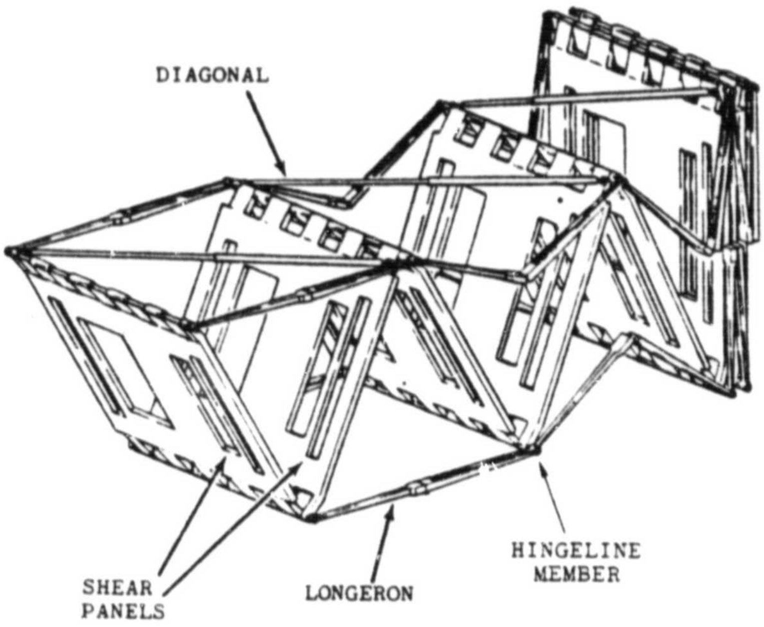


Figure 1.3-5. Concept 2—Warren Truss

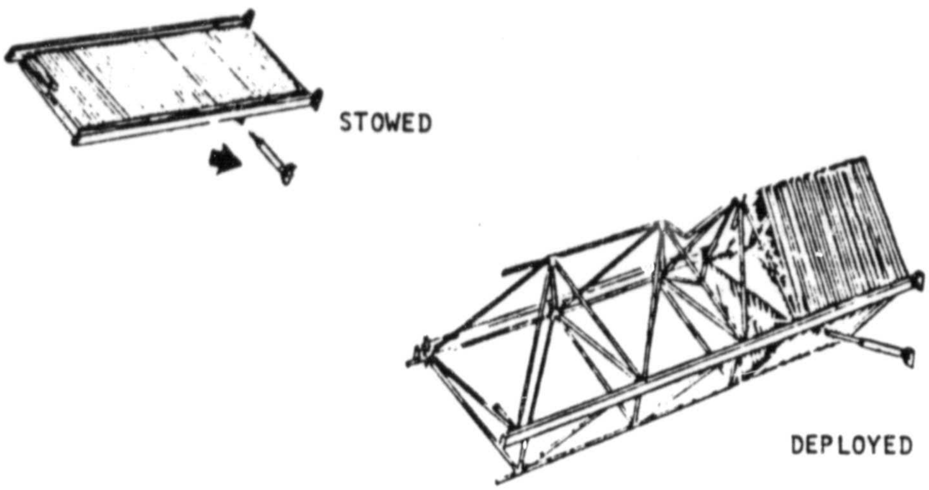


Figure 1.3-6. Concept 3—Tetrahedral Truss

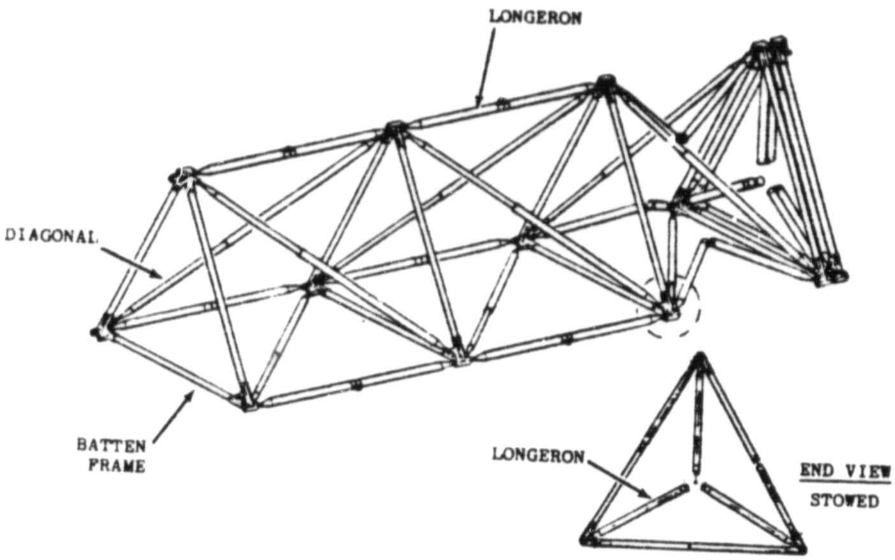


Figure 1.3-7. Concept 4—Truss

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- Concepts 5 and 7 (Figure 1.3-8) utilize the advantages of double folding (increased structure depth, fewer members and joints, and increased packaging efficiency) with provision of a rigid main housing structure. Both concepts utilize telescoping battens to deploy laterally to the larger cross-section shown. The increased depth provided by the double-fold permitted consideration of X-braced tension cables (Concept 5). However, in view of the ever present concern for maintaining cable tension throughout the thermal variation spectrum, a design with compression diagonals was considered (Concept 7).

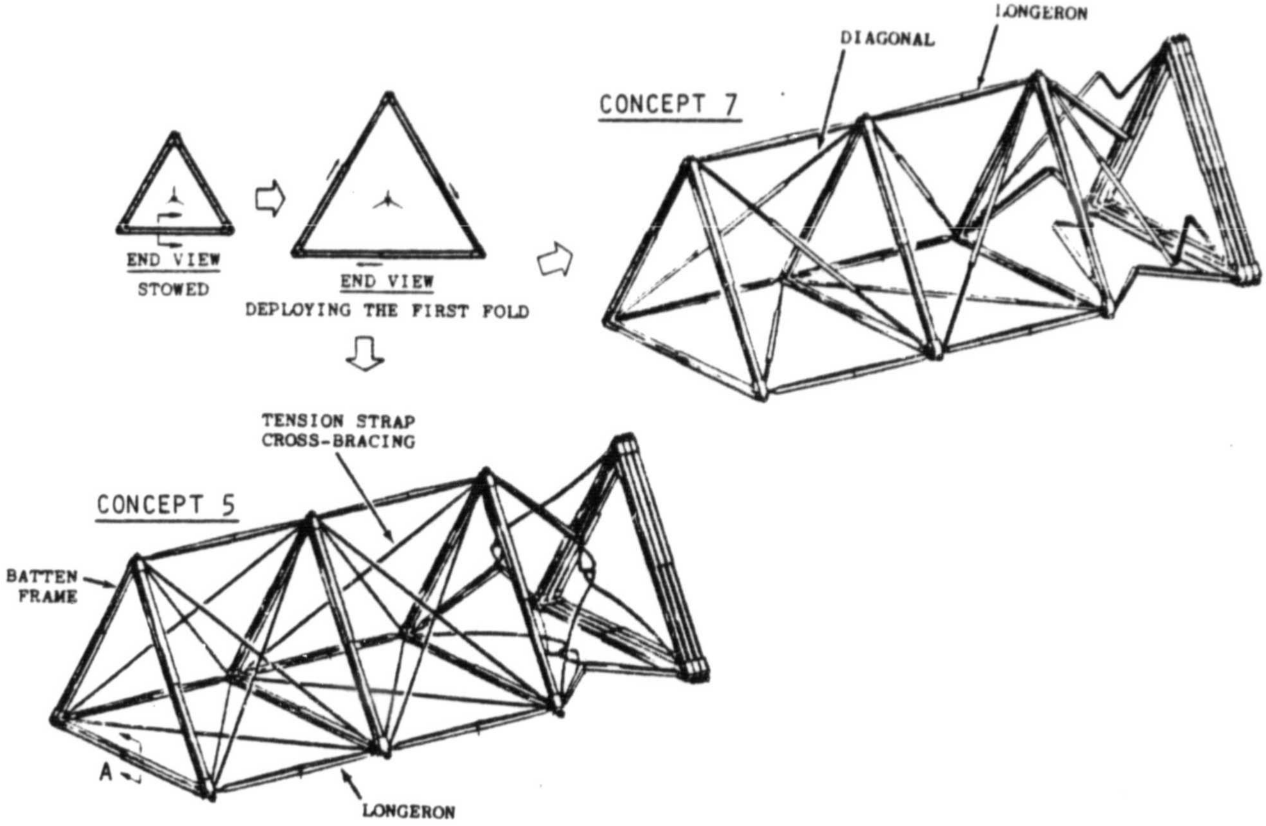

Figure 1.3-8. Concepts 5 and 7—Truss

- Concept 6 (Figure 1.3-9) establishes a square truss version of Concept 4, i.e., the same emphasis upon structural simplicity. The square truss can be either statically determinate (battens braced at end bays only) or redundant (all batten bays braced).

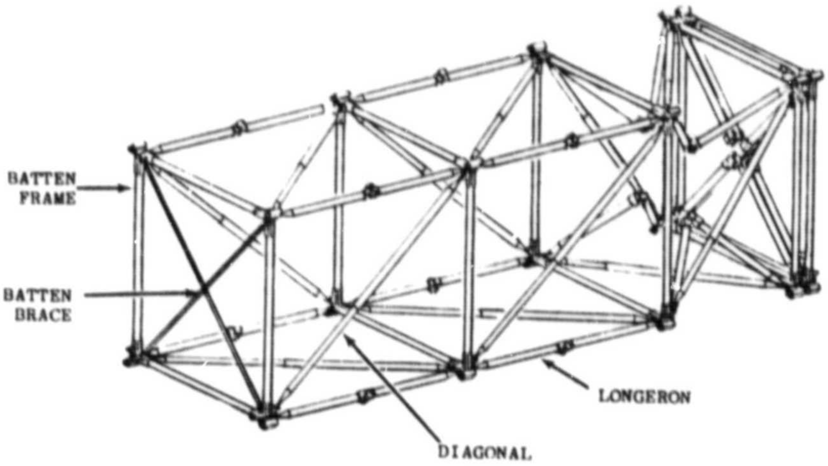

Figure 1.3-9. Concept 6—Truss

- Concept 8 (Figure 1.3-10) has the advantages of the square truss (redundancy, ease of building-block to building-block attachment, accommodation of payloads), but with the high packaging efficiency achieved through nesting of the longerons into the battens, and the availability of the total area inside the batten for mounting utilities in trays.

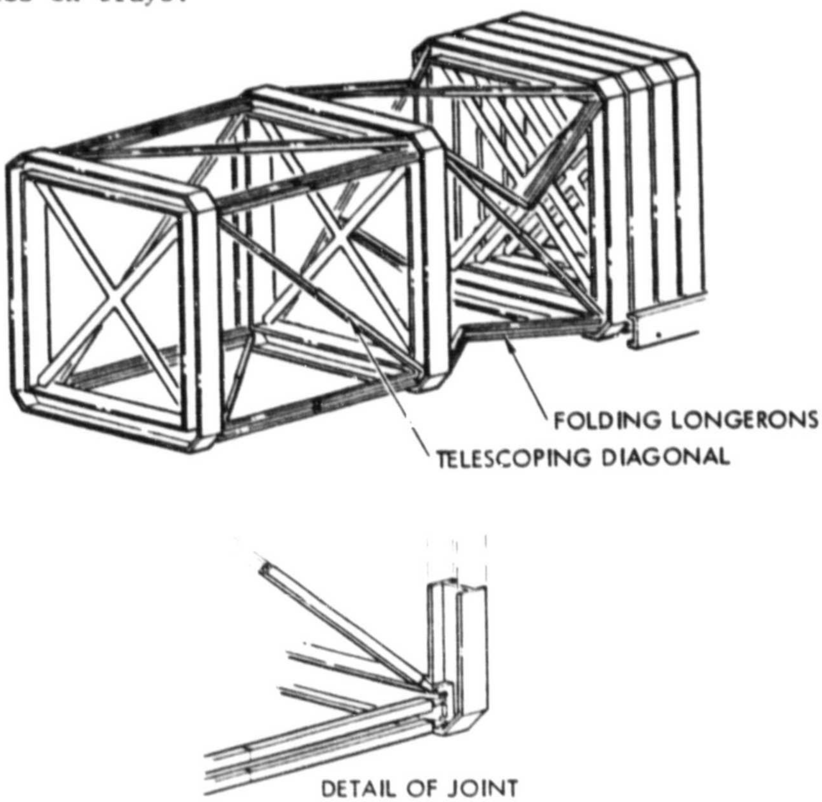

Figure 1.3-10. Concept 8—Truss

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1.3.2 Utilities Installations

1.3.2.1 Requirements/Objectives

The requirements/objectives for installation of the utilities into the deployable structure are as follows:

- o Incorporate the adopted requirements for power, data, signal and fluid lines established in Sections 1.1.2 and 1.1.3.
- o Growth capability is desirable.
- o Automatic deployment, as part of the building block, in space and on the ground with manual or ground support equipment (GSE) assistance if necessary.
- o Retraction in space is not necessary, but on the ground retraction is required with manual or GSE assistance if necessary.
- o Minimum number of joints/connections along a truss with no joints preferred.
- o Minimum number of in-space connections with none being preferable. This depends as much on the method of integrating the building blocks into the platform, as it does on integrating the utilities into the building block.
- o On the ground end-to-end checkout is highly desirable including all building blocks and possibly small payloads with no break/remake connections between ground checkout, loading in the orbiter, and final deployment.
- o Protection from adverse environments (thermal, radiation, vibration during launch)
- o High reliability
- o Weight - not a big driver for LEO but important for GEO.
- o Separation of power and data/signal lines (minimum electrical interference)
- o Accessibility - ease of installation, maintenance and replacement, and accommodations of design changes

1.3.2.2 Installation Methods

Four general methods for installation of utilities were investigated;

- In trays
- In coils or loops of several configurations
- On the outside of structural members; for example, secured to the outside of longerons
- Inside structural members such as the longerons

The results of a design review are shown in Table 1.3-2. The conclusions to be drawn from this review are:

- No one method of utilities installation is best for every configuration and mission.
- If room is available, the tray method is preferred for integration of a large quantity of utilities.
- For a limited number of utilities, installation on the outside of longerons is preferred.

1.3.2.3 Fluid lines

The fluid lines are nominally two centimeter diameter flexible lines containing Freon for cooling purposes. Other sizes and shapes are available to suit the specific requirements of the various configurations investigated. The two main problems encountered in using such lines are the bend radius and associated bending moment, and temperature control.

The bend radius required in any truss installation depends on the pitch distance between batten frames in the packaged configuration. The designer naturally attempts to keep this distance as small as possible to obtain a high packaging ratio, hence the importance of bend radius. If it is necessary to maintain the folded line in one plane the best bend radius obtainable is equal to the pitch distance less half the diameter (Figure 1.3.-11). If it is possible to overlap the lines out of plane, the bend radius can be increased (Figure 1.3-12).

The acceptable bend radius of a 2-centimeter-diameter flexible fluid line is 3 centimeters with an associated bending moment of 7 Nm (courtesy of Metal Bellows Corp.). In the event that a 2-centimeter-diameter line cannot be installed, there is always the alternative of using a greater quantity of smaller lines i.e. four 1.4 centimeters diameter lines. Another interesting possibility is the "race-track" shape (Figure 1.3-11) which permits a small bend radius. Table 1.3.-3 lists some characteristics of utilities installed on the Rockwell building blocks. In cases where the permissible bend radius of the installation falls below that which is recommended for a 2 centimeters diameter line, the system uses smaller lines.

Fluid lines require thermal insulation to prevent the Freon from freezing during eclipse periods. Six layers of MLI (multi-layer insulation) is estimated to be sufficient protection (Section 1.1.3). The MLI is retained on the outside of the flexible line by a loosely woven nylon jacket which will not interfere with the flexing of the line.

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Table 1.3-2. Comparison of Utilities Installation

METHOD	PRO	CON
TRAYS	• SIMPLE INSTALLATION—PERMITS STRUCTURAL ASSEMBLY, RIGGING, AND TESTING PRIOR TO UTILITIES INSTALLATION • ACCESSIBLE FOR DESIGN CHANGES • EASE OF MAINTENANCE & REPLACEMENT • GOOD GROWTH POTENTIAL • METEOROID IMPACT PROTECTION • SEPARATION OF POWER AND DATA • GOOD SUPPORT FOR LAUNCH • DISPERSED AGAINST METEOROID IMPACT • GOOD HEAT DISSIPATION FOR ELECTRIC POWER LINES	• EXTRA COST • EXTRA WEIGHT (GEO) • EXTRA SPACE
COILS & LOOPS	• SIMPLE INSTALLATION—PERMITS STRUCTURAL ASSEMBLY, RIGGING, AND TESTING PRIOR TO UTILITIES INSTALLATION • ACCESSIBLE FOR DESIGN CHANGES • EASE OF MAINTENANCE & REPLACEMENT • LIGHTWEIGHT, LOW-COST INSTALLATION • FAIRLY GOOD HEAT DISSIPATION FOR ELECTRIC POWER LINES • GOOD SEPARATION OF POWER AND DATA • LARGE BEND RADII OF LINES	• GROUND CHECKOUT—INCREASED FOLDING TIME • LIMITED GROWTH POTENTIAL • NO METEOROID IMPACT PROTECTION • NOT DISPERSED AGAINST METEOROID IMPACT • LAUNCH SUPPORT COULD BE PROBLEM
OUTSIDE OF LONGERONS	• SIMPLE INSTALLATION • ACCESSIBLE FOR DESIGN CHANGES • EASE OF MAINTENANCE & REPLACEMENT • GOOD SUPPORT FOR LAUNCH • GOOD HEAT DISSIPATION FOR ELECTRIC POWER LINES • LOW-COST, LIGHTWEIGHT INSTALLATION	• LIMITED NUMBER OF UTILITY LINES • MAY NOT BE ABLE TO SEPARATE POWER AND DATA • NO METEOROID IMPACT PROTECTION • NOT DISSPERSED AGAINST METEOROID IMPACT
INSIDE OF STRUCTURAL MEMBERS	• METEOROID IMPACT PROTECTION • GOOD SUPPORT FOR LAUNCH • LIGHTWEIGHT INSTALLATION	• SMALL BEND RADII OF LINES (NEW TECHNOLOGY) • MAY REQUIRE INCREASED LAY ANGLE (INCREASED WEIGHT & LOSSES) • DIFFICULT TO INSTALL—INSTALLATION DURING PIECE-BY-PIECE STRUC. ASSY. • NOT ACCESSIBLE FOR DESIGN CHANGES • DIFFICULTY OF MAINT. & REPLACEMENT • GROWTH POTENTIAL POOR • MAY NOT BE ABLE TO SEPARATE DATA AND POWER • NOT DISPERSED AGAINST METEOROID IMPACT • POOR HEAT DISSIPATION FOR ELECTRIC POWER LINES • NOT SUITABLE FOR SMALL MEMBERS (SMALL PLATFORMS)

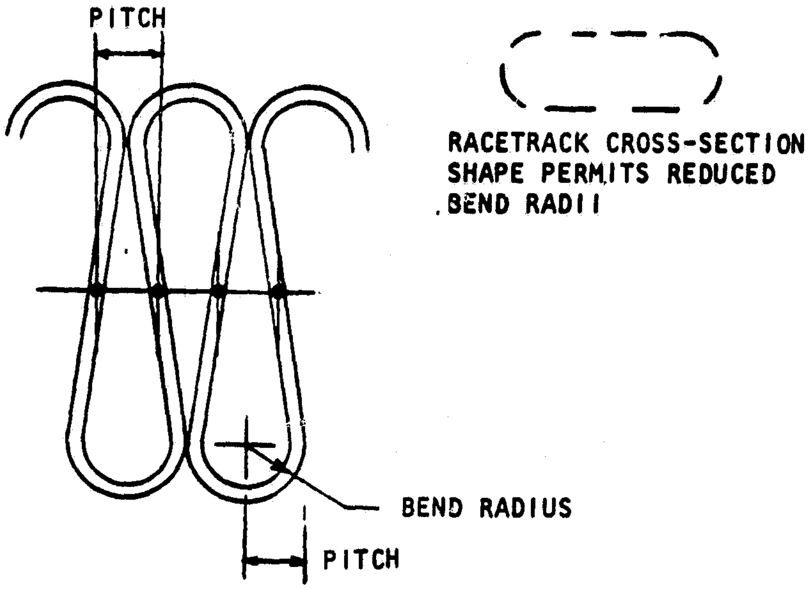

Figure 1.3-11. Utilities in One Plane

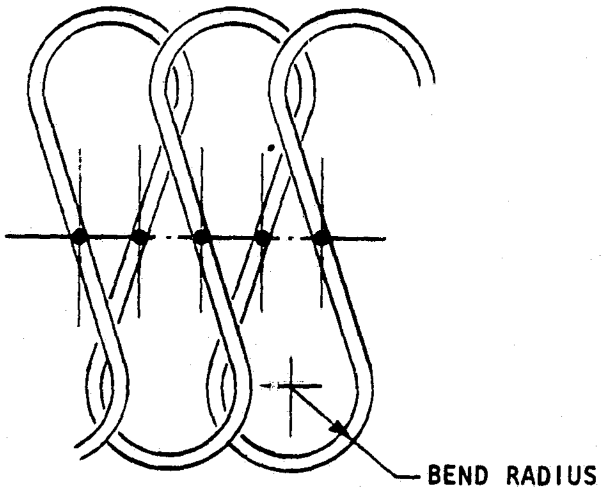

Figure 1.3-12. Utilities with Overlap and Increased Bend Radius

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Table 1.3-3. Electrical and Fluid Utilities Characteristics

CONCEPT	ELECTRICAL METHOD	ELECTRICAL BEND RADIUS (cm)	FLUID METHOD	FLUID BEND RADIUS (cm)
1	TRAYS	6.3	TRAYS	12
2	TRAYS	3.3	OVERLAP FOLDS	20
3	COILS	24.0	OVERLAP FOLDS	20
4	OVERLAP FOLDS	22.3	OVERLAP FOLDS	14
5	TRAYS	2.9	TRAYS	4
6	OVERLAP FOLDS	22.3	OVERLAP FOLDS	14
7	TRAYS	2.9	TRAYS	4
8	TRAYS	2.5	TRAYS	3.1
8	LONGERONS	20.3	LONGERONS	20

1.3.2.4 Electrical Utilities

The electrical utilities consist of power lines; twisted shielded pairs for both signal and data lines; and for data lines, fiber optics and coaxial.

The twisted shielded pairs and coaxial lines are of small diameter and are flexible and pose no problems pertinent to folding in the trays or into loops as required. The type of overall platform design envisioned by Rockwell avoids the problems of multiple coaxial connectors. There are no coaxial connectors on the trusses where flexing occurs and, since the entire platform is packaged and deployed as a unit without piece-by-piece assembly, there are very few connectors in a line which may run from one end of the platform to the other, traversing several building blocks in the process.

Fiber optics are sensitive to radiation degradation and to cracking caused by thermal cycling. Also, at low temperature some fiber optics cease to transmit. The present drive in the industry is to improve fiber optic materials and to develop shielding to overcome these problems. This is listed as a line item in the technology development section. In the event fiber optics are not ready for 1986, copper lines can be used instead.

Power lines for space platforms do not need the heavy insulation commonly found on electrical cables for earth applications. There is no moisture problem in space and no requirement for "idiot proof ruggedness". Consequently, a light, loosely woven insulation is recommended. Such insulation will save weight, enhance flexibility, and avoid cracking when folded.

The acceptable bend D/d ratios associated with the sizes of power cables selected are shown in Table 1.3-4.

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Table 1.3-4. Permissible Electrical Conductor D/d Bend Ratios

CABLE SIZE	D/d	BEND RADIUS (cm)
0	6	2.45
2	6	1.96
4	6	1.55
WIRE BUNDLES	10	23

The actual bend D/d ratios (Table 1.3-3) all exceed these acceptable values.

The permissible D/d shown in Table 1.3-4 were obtained by review of the parametric data (Figures 1.3-13 through 1.3-16) furnished by Tension Member Technology (TMT). The review used a maximum permissible strain of 2.5% based on upward adjustment of the strain data shown in Figure 1.3-18 since the data shown are for cyclic reversal of strain which does not occur in the bending of the cables. Further, the value of 2.5% is regarded as reasonable since the following analysis includes the strain imposed during fabrication of the cable which occurs only once. The basis of the 40 cycles of strain shown in Figure 1.3-17 comes from the assumption of 10 cycles of folding (during installation, checkout, and final deployment) multiplied by a scatter factor of 4.

The analyses are all based upon the use of a 15-degree lay angle (essentially standard cable fabrication practice).

- For a No. 2 conductor a maximum strain of 1.5% is determined from Figure 1.3-13 for D/d = 6. The D/d = 6 is conservative.
- For a No. 0 conductor a maximum strain of 1.1% is determined from Figure 1.3-14 for D/d = 6. The D/d = 6 is conservative.
- For a 1 x 7 wire bundle comprised of No. 2 conductors, from Figure 1.3-15, the curvature ratio for bending to a D/d = 10 is 19, the initial wrap curvature is 32. From Figure 1.3-13 the maximum possible strains are respectively 1.2 and 1.15%, or a total of 2.35%. A 1 x 7 bundle of No. 0 conductors is less critical. It is pertinent to note the cyclic strain is less than half of the total strain.
- A 1 x 19 wire bundle to a D/d = 10 will have less strain.

The bending moments associated with bending these cables, to the D/d ratios discussed herein, are not significant to the concept development. For example, the limit bending moment incurred in bending a No. 0 cable to a D/d = 6 is 0.7 Nm.

The background for the foregoing analysis is based upon the additional documentation (and discussions) provided by TMT as follows:

The bending of a helically wound cable produces bending, extensional, and torsional strains in the individual cable elements. The bending strain is a result of the change of curvature of the elements as the cable is bent.

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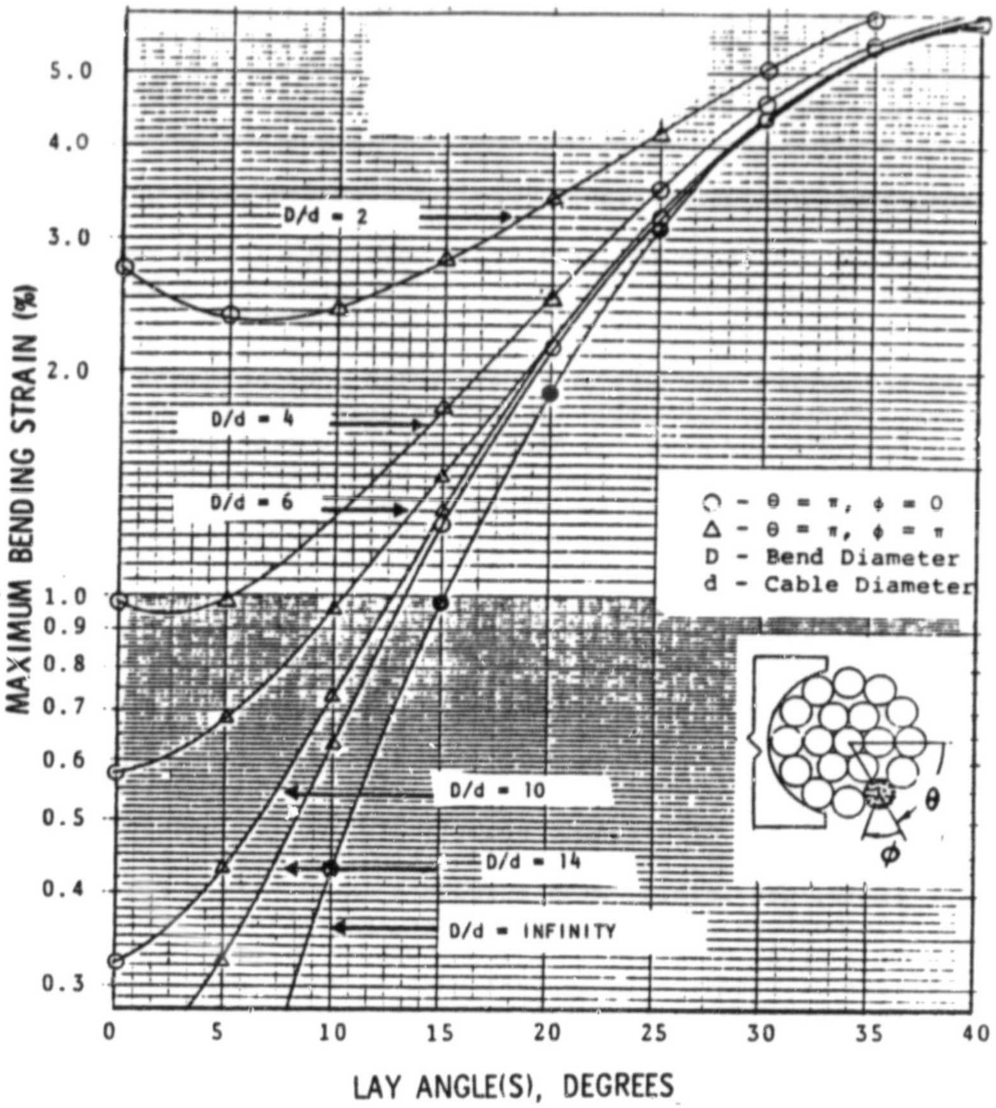

Figure 1.3-13. Maximum Bending Strain —No. 2 Conductor

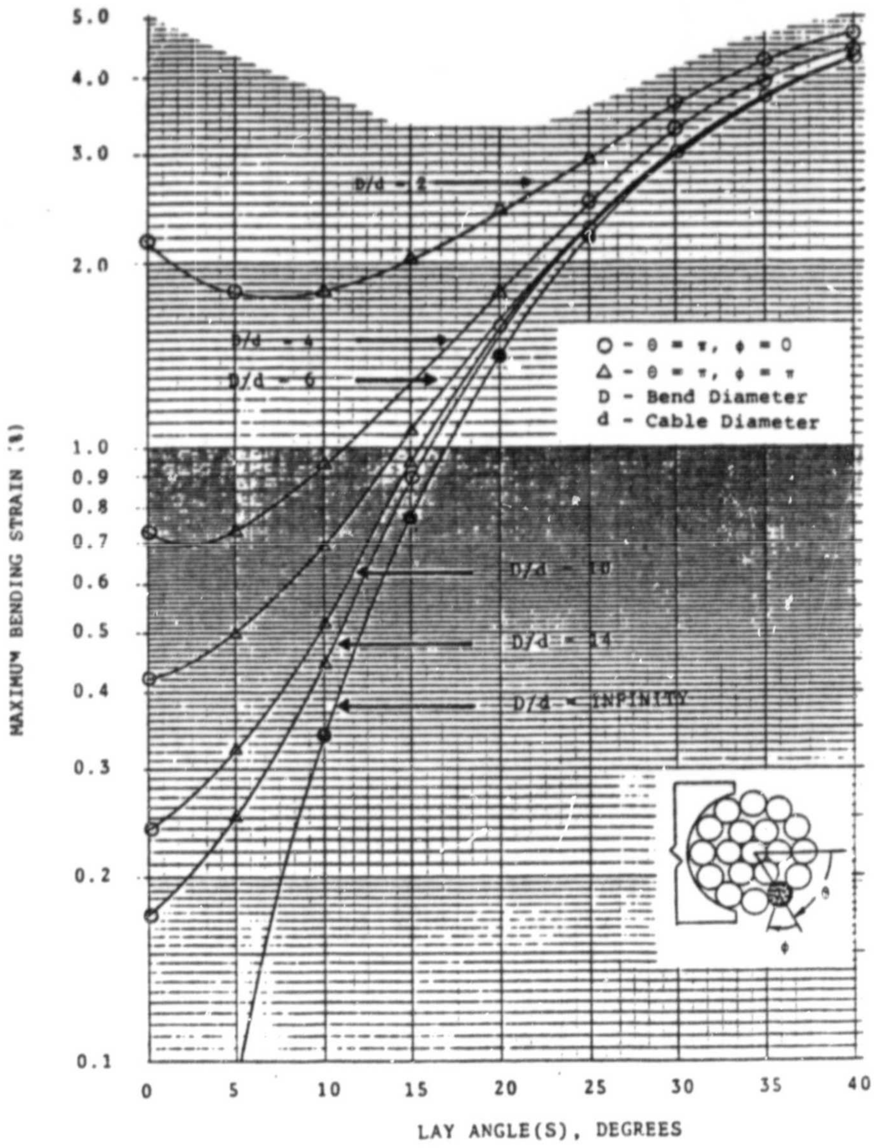

Figure 1.3-14. Maximum Bending Strain —No. 0 Conductor

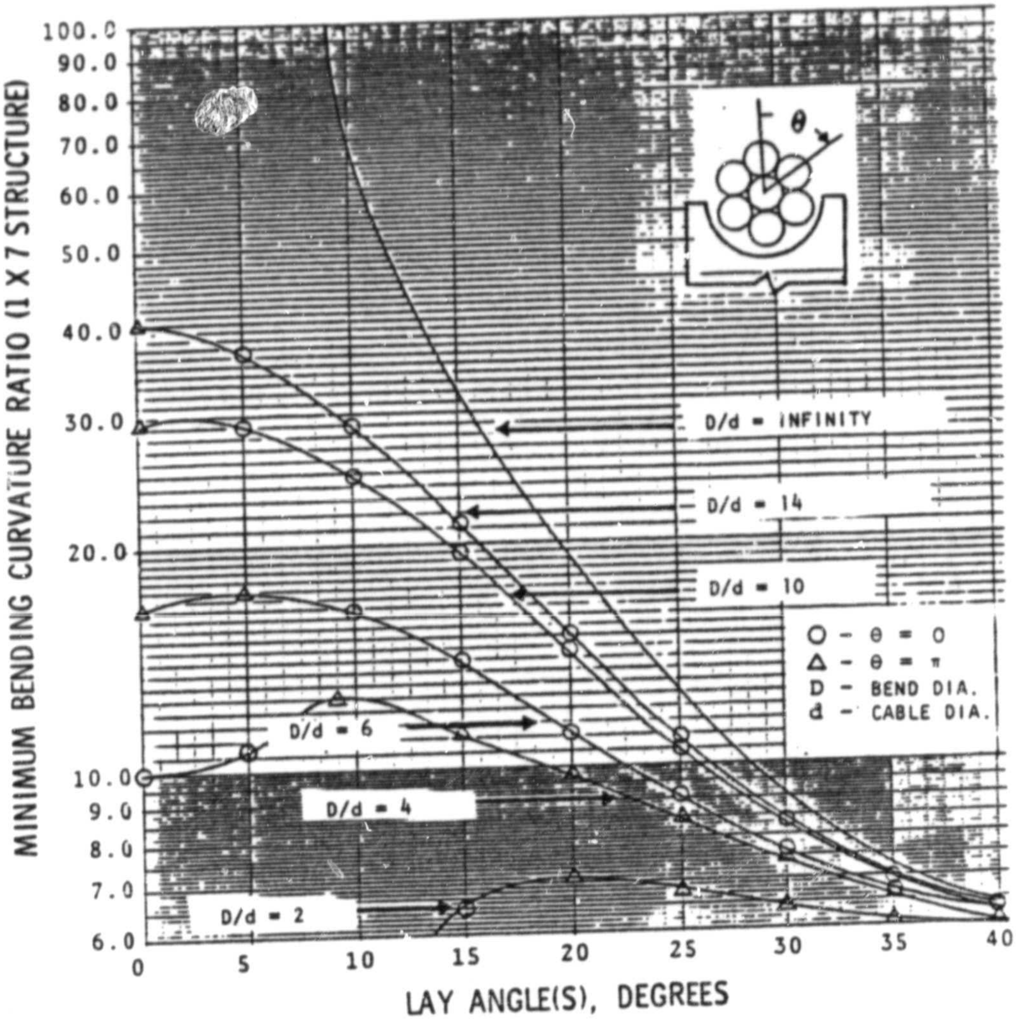

Figure 1.3-15. Minimum Bending Curvature Ratio —1 x 7 Structure

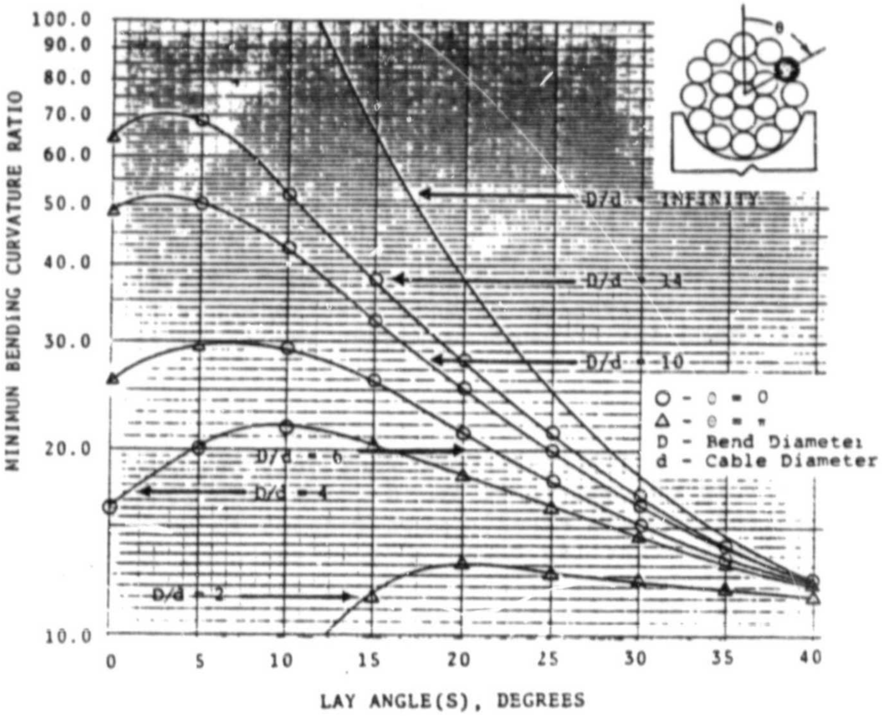

Figure 1.3-16. Minimum Bending Curvature Ratio —1 x 19 Structure

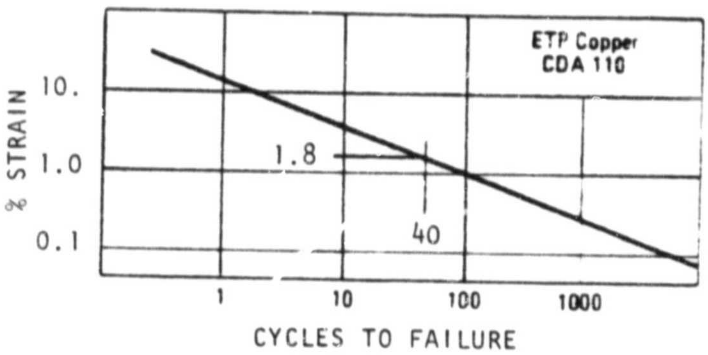

Figure 1.3-17. Copper Filament Fatigue Data

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Extensional strain occurs when the elements of the cable are prevented from axial movement due to friction forces among them, thus requiring the elements to elongate or foreshorten to comply with changes in their local path length as the cable is bent. This friction-induced extensional strain of the cable elements is in addition to any strain which may exist due to tensile loading and elongation of the complete cable assembly. Torsional strains are the result of twisting of the individual elements due to bending distortions of their helical paths.

Both the bending and torsional strains induced in individual cable elements become smaller for decreasing element size relative to the total cable size. Extensional strains due to bending become smaller with reduced values of internal cable friction.

The analysis described in this report assumes that both the extensional and torsional strains are negligible and that only bending strains are significant. This assumption is valid for a frictionless cable wherein the elements are free to move axially with respect to each other. The known techniques for approximating this frictionless condition are the utilization of ample cable lubrication and/or a "loose pack" cable design. For some applications, low cable friction can be achieved by enclosing the cable assembly within a loose-fitting tube or hose which contains a lubricant such as a liquid fluoropolymer. Of course, such a design is not suited for deployable space structures. However, a "loose pack" cable design is quite appropriate. In such a cable, the individual elements ideally are helically wound and held loosely together with some average spacing among them. The result is a nearly frictionless cable which provides for minimum strain on the individual elements during cable bending.

The No. 2 (19 x 35) and No. 0 (19 x 55) cable constructions are defined as "cables" which consist of 19 "strands," each of which consists of a number of "elements" (35 and 55, respectively). The 19 strands are arranged with one strand in the center of the assembly, a layer of six strands around the center strand, and a layer of twelve strands around that. The 1 x 7 and 1 x 19 constructions are defined as "cables" (or "strands") which consist of a number of "elements" (7 and 19, respectively). In each case, the "elements" are the smallest subunits of the construction.

The numerical analysis included computation of the bending strains induced in the individual cable elements both during original manufacture of the cable assembly and as a result of bending this assembly to various ratios of bending diameter to cable diameter. Figures 1.3-13 and -14 show the results of this analysis for the No. 2 and No. 0 cable constructions. For this analysis, it was assumed that the helical direction of the elements within the individual strands was opposite that of the strands in the complete cable assembly. It was also assumed that the 19 strands were manufactured as a "bunched" strand configuration. Furthermore, the analysis required that the lay angle of the elements within the strands be equal to the lay angle of the strands within the cable assembly. Without this restriction, the analysis would have required an additional plotting dimension.

The curves in Figures 1.3-13 and -14 are dimensionless and can be applied to any size cable made with the indicated configurations. On each of these curves, the reference angle Θ defines the location of the strand within the cable (Θ = 0 corresponds to the strand furthest from the center of curvature of the entire cable assembly). Similarly, the reference angle Ø defines the location of an individual element within that strand (Ø = 0 corresponds to the element which is furthest from the cable centerline). The combinations of Θ and Ø indicated in these figures correspond to the locations of the maximum element bending strain (minimum radius of curvature).

The curve for D/d = infinity corresponds to a straight cable assembly and indicates the maximum bending strain (minimum radius of curvature) in individual elements as the result of the elements being formed into a double helix during the cable manufacturing process. The location of this maximum bending strain corresponds to Ø = 0 for all values of Θ. In other words, for a straight cable, this maximum bending strain occures in the cable elements which are furthest from the cable centerline.

The remaining curves for various values of D/d indicate the maximum bending strain (minimum radius of curvature) in the individual elements after bending the entire cable assembly. In all cases, this maximum bending strain is that which is produced as a cable element, which is initially straight, and assumes some final radius of curvature in the bent cable assembly. Not included in this analysis is any initial state of strain which the individual elements may have had as the result of wire drawing or heat treating processes occurring prior to the elements being formed into the cable.

Note that the location of the maximum bending strain changes as a function of both the element lay angle and the ratio of the bending diameter to the cable diameter. The reader is cautioned against attempting to take the difference between the D/d = infinity curve and any other curve to determine the change in element bending strain due to bending the entire cable assembly. This procedure will not yield accurate results in all cases, since the site of maximum bending strain within a straight cable may be different than the site of maximum bending strain within a bent cable. Furthermore, even if the sites of maximum bending strain are identical in both the straight and bent cable, it is possible that for some cable geometries and bending diameters, the radius of curvature of an individual element may pass through infinity as the cable is bent, thereby producing a change in strain due to bending which is greater than the net bending strain which exists either before or after the cable is bent.

Figures 1.3-15 and -16 for the 1 x 7 and 1 x 19 configurations indicate the minimum bending curvature ratio, which is the minimum radius of curvature of an outer layer element divided by the radius of that element. Again, these curves are dimensionless, and they can be applied to any size cable made with the indicated configuration. The curve for D/d = infinity can be used to describe the minimum radius of curvature of one of the outer elements in a straight cable. The remaining curves indicate the minimum radius of curvature of one of the outer cable elements after the cable is bent.

All the four curves have been corrected to take into account the fact that the diameter of the complete cable assembly increases with increasing lay angle of the individual elements and strands. In other words, for a D/d ratio of 10, a larger bending diameter is required for a cable assembled with 20-degree lay angles than is required for a cable assembled with 10-degree lay angles.

The curves in 1.3-13 and -14 may be used directly to determine the maximum bending strain produced in an individual element of a No. 0 or a No. 2 conductor as the result of the original cable manufacturing process or as a result of bending the cable to a specified bending diameter. It is important to note, however, that when establishing a value for D/d, the "d" applies to the outside diameter of a bare conductor. Any insulating jacket which is applied over the conductor must be ignored for purposes of determining the maximum strain.

All the curves require that the lay angle of the cable component be specified in order to determine the maximum bending strain or the minimum bending curvature ratio. For a cable component which follows a simple helical path, the lay angle is defined as:

$$\text{Lay angle} = \text{Arctan}\ \frac{2\pi r}{\ell}$$

where

r = the pitch radius of the cable component as measured from the axis of the helix to the center of that component, and

ℓ = the lay length of the cable component (the distance measured along the axis of the helix corresponding to one helical pitch of the cable component).

1.3.3 Deployment Mechanization Concepts

The requirements for the deployment mechanism are:

- Automatic deployment - in space, on the ground manual or GSE assistance is permissible.
- Retraction is nice but not a firm requirement.
- One bay at a time deployment.
- Controlled rate of deployment. Sometimes it may be necessary to synchronize several trusses being deployed.
- Root strength of truss maintained throughout deployment.
- Suitable for use with the building block approach for deployment of a platform.
- Compact or foldable.

- o Low power consumption.
- o High reliability.
- o Suitable and safe for EVA operations in the event of a malfunction.
- o Able to generate extra force in the event of a "hang-up" or jam.

The following sections describe the deployment techniques considered in this study.

1.3.3.1 Pressurization

Pressurization systems were investigated but no suitable applications were discovered.

1.3.3.2 Cable Deployment Systems

Cable systems are sometimes used for deployment and/or retraction of trusses and similar structures. They may be used as the prime motive source or in conjunction with deployment springs, in which case the cable may function as a restraining device. Cable systems tend to deploy all the bays of a truss simultaneously or in a random fashion. This may be acceptable for a single truss but is not acceptable for a system with many trusses. A remedy for this uncontrolled deployment is to tie all of the batten frames together by latches which are released sequentially. Alternatively a "hold-back" sequencing mechanism can be incorporated in the main housing. Another and more complex method is to have a separate cable system for each bay.

Depending on the truss being used, a cable system will probably not develop root strength of the truss unless it is aided by auxiliary guide rails.

Although cable systems were investigated, the truss concepts developed herein were not appropriate for the use of cables.

1.3.3.3 Stored Energy Deployment Systems

Of the stored energy devices available, mechanical springs are the most suitable for truss deployment. Other devices such as gas cylinders are more complex and bulky and less relilable. Mechanical springs used may be tension, torsion, compression, leaf or any of the other forms available.

Spring deployment systems have many of the same characteristics as cable systems; simultaneous bay deployment, lack of accurate control over deployment rate, and lack of root strength development. These drawbacks can be overcome by the addition of other devices, such as restraining cables, sequenced latches and guide rails.

Concepts 4 and 6 were initially designed for torsion spring deployment using sequenced mechanisms as shown in Figure 1.3-18. This is a mechanism which holds adjacent battens together as part of the truss stowed stack. Upon receipt of a command signal, a fuse wire which holds together the two halves of a nut is melted. The nut and bolt separate and the batten is free to deploy under the influence of torsion springs (not shown).

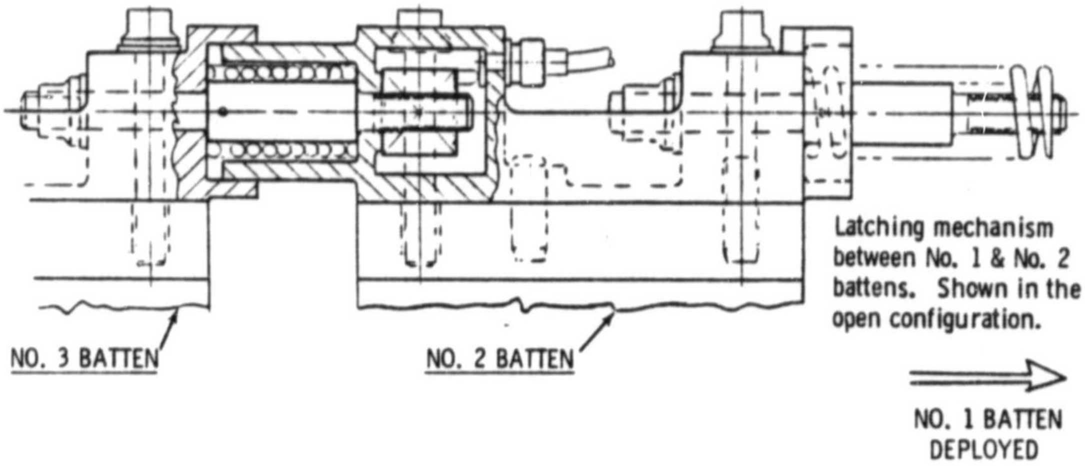


Figure 1.3-18. Latching Mechanisms between No. 2 and No. 3 Battens

Spring deployment is usually not as positive as a mechanical drive. It is sometimes difficult to obtain a reserve of force without inducing unwelcome accelerations in the item being deployed. For longerons which unfold and move to an over center or "in line" configuration, light springs located at the longeron center hinge are very useful. They can provide a kick over the last few degrees of movement which is difficult to obtain by other means. Such springs were used in the design of longeron latches as shown in Figure 1.3-19.

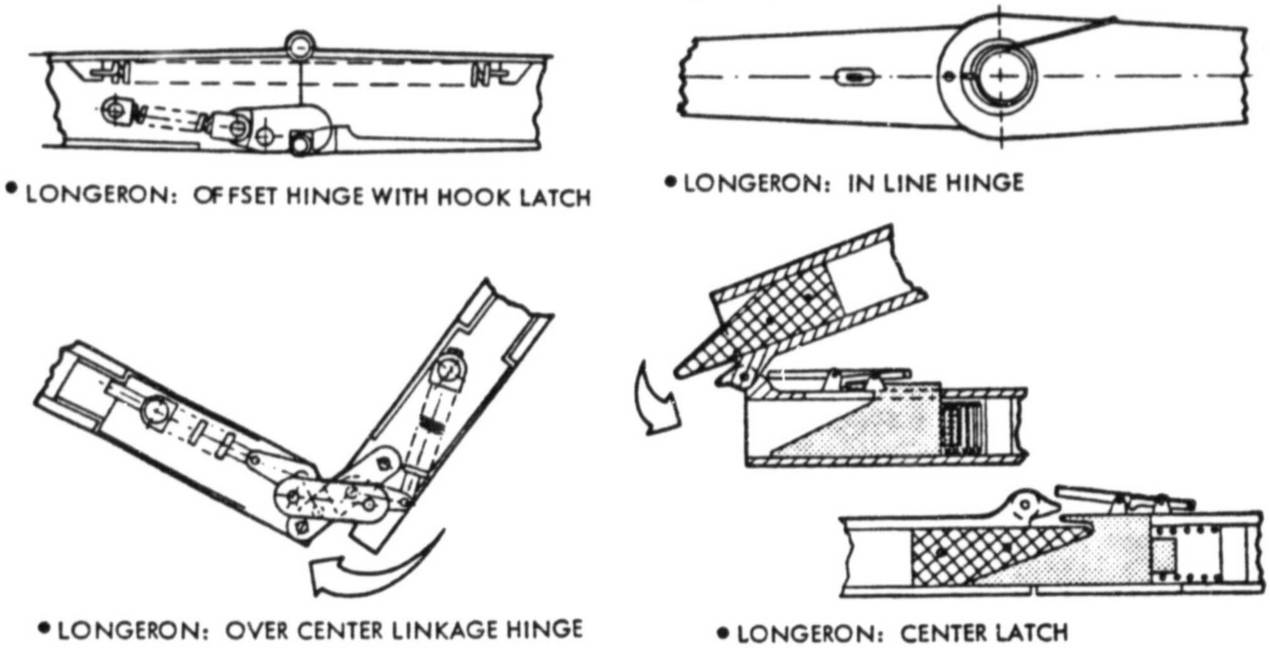


Figure 1.3-19. Longeron Latching Mechanisms

1.3.3.4 Mechanical Deployment System

The mechanical deployment system used is the same or equivalent to the reciprocating tape and pulley with integral guide rails developed by General Dynamics for their tetrahedral truss (Figure 1.3-20).

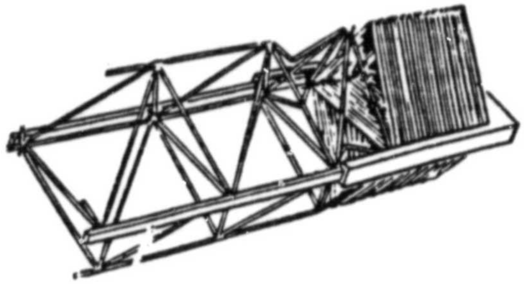


Figure 1.3-20. Deployment Rails and Mechanism

The system consists of two guide rails (Figure 1.3-21), each carrying a tape and pulley arrangement which advances to deploy one bay, then returns to deploy the next bay. To completely control the motion of the truss and to develop root strength, the guide rails need to extend a distance of two deployed bays from the front of the stowed truss stack. The mechanism needs to extend only half way along that distance. The pulleys are motor driven and are controlled electronically. A method of folding the guide rails and mechanism is shown on Drawing 42712-016, sheet 4 (Volume II).

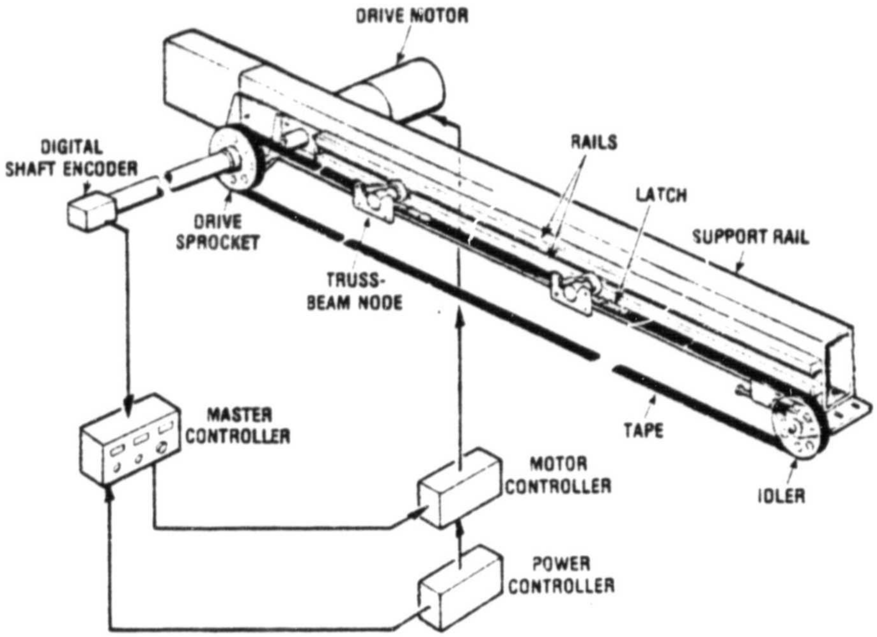


Figure 1.3-21. Detail of Deployment Rail

For a truss whose profile changes during deployment (Concepts 1, 2 and 3) the two rail system just described is used. However, for trusses whose profile does not change during deployment (Concepts 4, 5, 6, 7 and 8) a variation is available. It is possible to use 3 or 4 guide rails, (depending on whether the truss is triangular or square) the length of the rails being equal to only one bay in front of the stowed truss stack. The operation of the rails and mechanism remains essentially unchanged.

The reciprocating mechanical deployment system is the design selected for the deployable platform. It meets all of the requirements and does not suffer from the drawbacks of the other systems considered.

1.3.3.5 Payload Deployment

One problem associated with the use of guide rails arises when extending a truss which has a payload or module so wide that the guide rails cannot straddle it, (Figure 1.3-22) and therefore cannot be unfolded until the truss has extended and moved the large payload out of the way. Obviously, if the guide rails are not in position when the truss/payload is moving, there is no root strength developed.

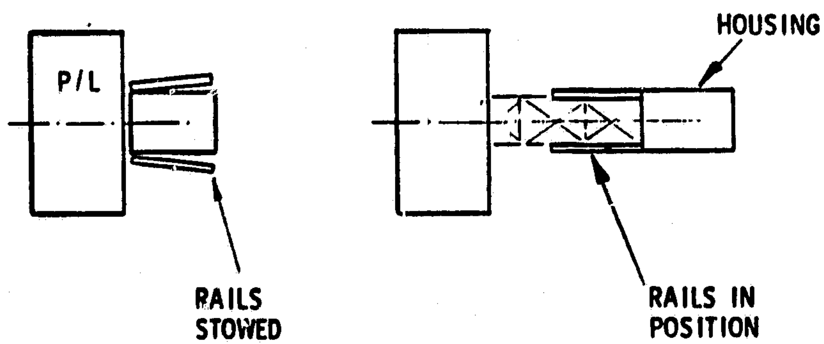


Figure 1.3-22. Deployment Rail Issue

There are several possible solutions.

Solution 1: Use fixed rails (Figure 1.3-23). This design uses more stowage length in the orbiter and the large payload is not supported directly from the main housing.

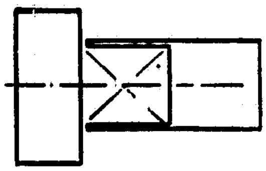


Figure 1.3-23. Fixed Rail Method

Solution 2: Penetrate the module (Figure 1.3-24). It is not likely that the rails would be allowed to penetrate an actual payload but it is quite possible that they could penetrate a structural module.

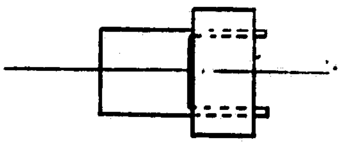


Figure 1.3-24. Penetration Method

Solution 3: Spar booms (Figure 1.3-25). This appears to be the most feasible solution to the stated problem for situations with payloads and modules as well.

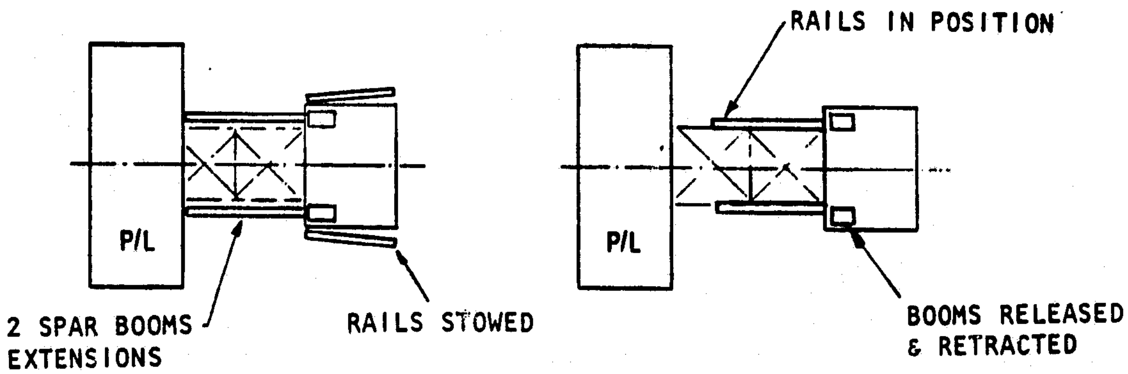


Figure 1.3-25. Spar Boom Method

1.3.4 Main Housings and Adapters

This section describes the array of main housings and adapter designs (Figure 1.3-26) developed for the candidate concept trusses (discussed in Section 1.3.1) that are compatible with the building-block concept. Additional information is provided in the drawings listed in Volume II.

Emphasis in the drawing development was placed upon conceptual rather than detailed design. Of primary concern was the overall concept integration with the truss, rail system, accommodation of docking ports and payloads, and suitability for inter-building-block attachments. Sufficient detail was presented to support the weights and cost analyses (Section 1.4).

The primary functions of the housings are as follows:

- Attachment of the deployable truss, mechanization, and rail system components.
- Attachment of electrical, data, and fluid utilities feed-through connections (as required).
- Provisions for mounting of docking port support rings for orbiter berthing or payload attachment.
- Provision for structural attachment with adjacent building-block housings and/or adapters with accommodation of the variations in the building blocks orientation.
- Provision for structural attachments for orbiter installation.

The structural concept for all the main housings shown is expected to be built up from numerically controlled integrally machined aluminum panels (2219-T6 or equivalent) to a truss or skin-stiffened construction depending on the specific design conditions. Thermal gradients can be greatly minimized to reduce the local thermal distortion (if required). If aluminum is not adequate, (depending on the pointing accuracy requirement) the main housing can be constructed of composite materials, but with increased cost.

The adapter functions are as follows:

- Provision of attachments to mount onto the extremity of the basic deployable truss.
- Provision of all hardware to permit subsequent RMS attachment of payloads, RCS modules, and/or orbiter berthing ports.
- Provision of automatic electrical and fluid line connectors.
- Provision of structural stability to the main housing during orbiter boost.

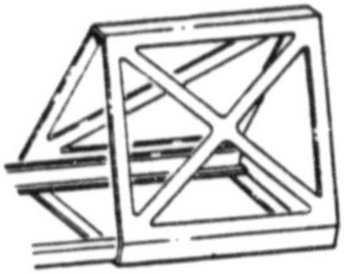
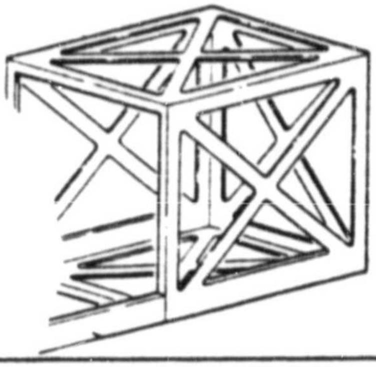
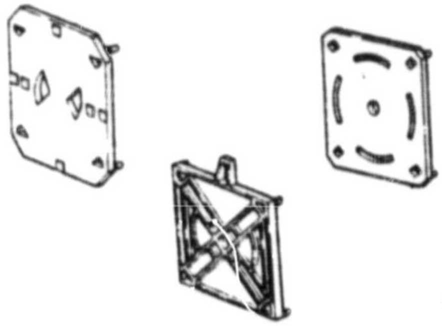
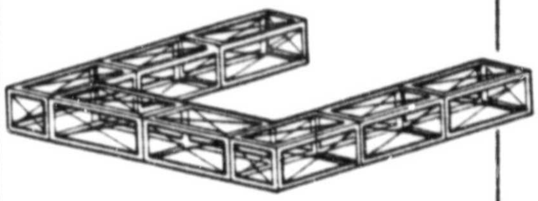
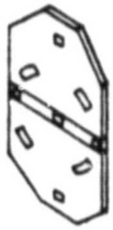
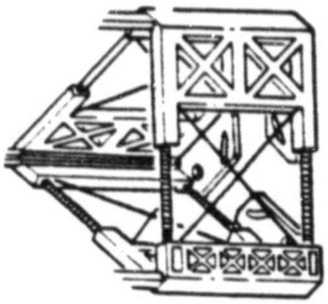
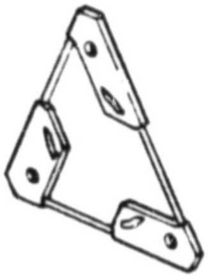
TRUSS CONCEPT	HOUSING CONCEPT	ADAPTER CONCEPT
1 4		
2 6 8		
3		
5 7		

Figure 1.3-26. Candidate Housing and Adapter Concepts

While three concepts are shown for the adapter, the most likely construction is that of a numerically controlled, integrally machined aluminum design (thermal stability permitting). The machined panel permits the greatest flexibility for mounting of latches, connectors, etc.

1.4 DEPLOYABLE PLATFORM SYSTEMS CONCEPT INTEGRATION

This section describes the integration of the specific deployable truss concepts (1.3.1), utility folding concepts (1.3.2), deployment mechanization concepts (1.3.3) and main housings and adapters (1.3.4) into 8 candidate building block concepts that are subsequently compared in the concept selection study of Section 4.

Unquestionably, a comparison of these diverse designs for satisfaction of a specific platforms unique configuration and performance requirements represents an ample challenge but is possible within the scope of this study. Comparison of the 8 concepts for more than one platform of different size, strength, stiffness, and utilities accommodations needs, while preferable, was beyond the scope of this study. The best compromise therefore, was to perform the comparison in detail (production of drawings) for one baseline platform designed to one set of baseline requirements and, to the maximum extent possible, analyze and/or review the implications associated with departure of size and requirements from the baseline.

The generic platform described in Section 1.2 was the baseline platform, with the adopted strength, stiffness, and utilities accommodations representing the baseline requirements. All 8 concepts were constrained in size to permit packaging of the generic platform in the orbiter as schematically shown in Figure 1.4-1. Additional details pertinent to packaging are shown on Drawing 42712-020 (Volume II). The resulting deployable truss dimensions are shown on Figure 1.4-2. It is recognized that other options for each concept are possible. The options shown were the most likely at that point in the study. In fact, Concepts 2, 6, and 8 were subsequently (after completion of structural and thermal analyses) packaged as shown in Figure 1.4-3, permitting an increase in the truss width and depth. Hence, the data shown in Section 4 for these concepts are slightly pessimistic.

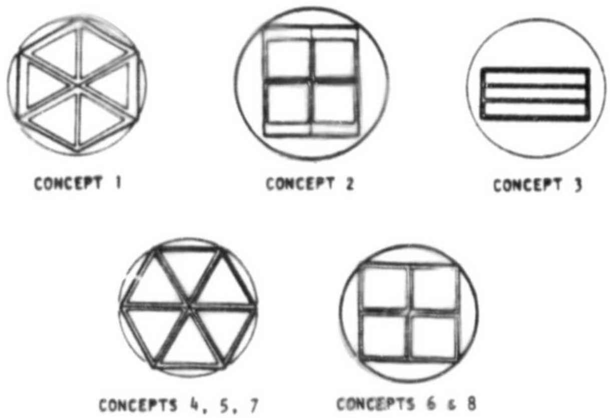


Figure 1.4-1. Concepts for Packaging of Generic Platform

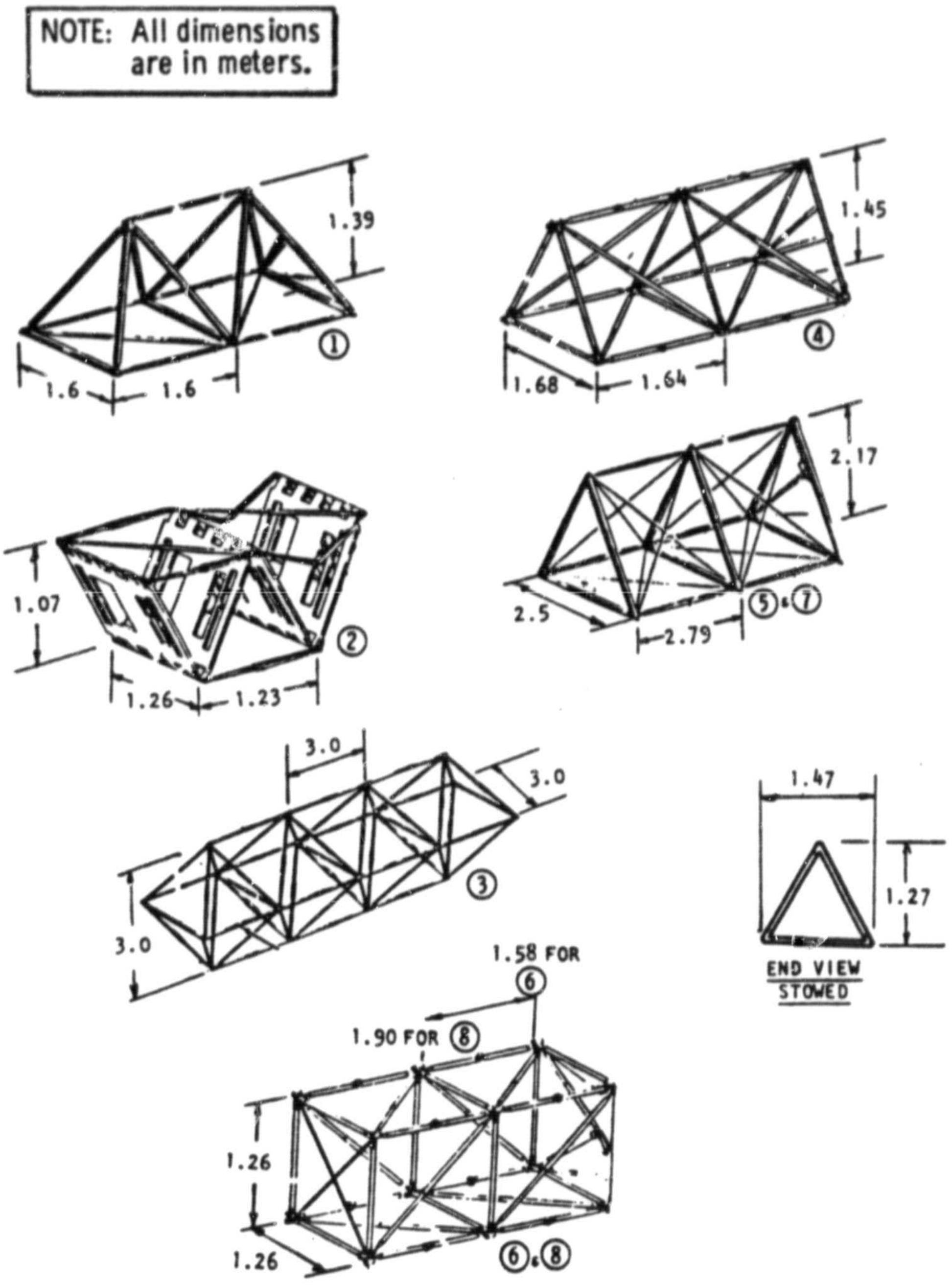


Figure 1.4-2. Structural Concept Dimensions

Figure 1.4-3. Final Concept for Packaging (Concepts 2, 6, and 8)

Referring to Figure 1.4-2, it is pertinent to note that the largest truss dimensions compatible with orbiter space available, were used for the following reasons:

- o Packaging efficiency is increased with increase in truss depth
- Best accommodation of utilities
- o Fewest structural members and joints (reduced cost and minimum weight)

The following sections describe the structural sizing method and approach, the 8 integrated candidate building-block designs, the packaging of the generic platform utilizing these designs, and the thermal, mass properties, and fabrication cost data developed for use in the concept selection. Finally, a discussion of significant miscellaneous issues is presented.

1.4.1 Structural Sizing Method and Approach

A summary of the design ultimate compression load, shape, and structural sizes for the 8 deployable truss individual longeron, batten, diagonal and/or pyramidal members is presented in Table 1.4-1. The generation of these data are described as follows:

- o The individual compression loads were obtained by computer analysis of each truss (dimensions shown in Fig. 1.4-2) for the concurrent adopted limit bending and torsional moments of respectively, 2.5×10^4 and 1×10^4 Nm. An ultimate safety factor of 1.5 was used. The computer used equations developed from hand analyses which are sufficiently accurate for the purposes of this study.

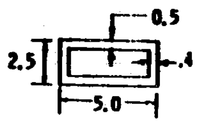
- o The individual compression load in conjunction with the member length and specified material properties was input (automatically) into a column analyses subroutine. A CRT plot for each member is obtained such as that shown in Figure 1.4-4 for either longeron, batten, diagonal, or pyramidal members of circular, square, rectangular or I shape. The material properties used for the T300/934 graphite composite design was E_L=143,000 and E_T = 17,250 mpa and for the P75S/934 E_L = 231,000 and E_T = 28,000 mpa.

Table 1.4-1. *Structural Sizes to Adopted Strength and Stiffness Requirements*

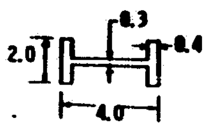
CONCEPT	CHARACTERISTIC	LONGERON	PYRAMIDAL	DIAGONAL	BATTEN **
1	ULT. LOAD (KN)	28.8 ✓	12.0	15.3	5.4
	SHAPE	R	I	C	I
	DIMENSIONS (CM)	5.0, 0.43, 2.5, 0.43	5.5, 0.1, 5.3, 0.18	4.7, 0.15	5.4, 0.09, 4.6, 0.13
2*	ULT. LOAD (KN)	24.3 ✓	—	19.6	—
	SHAPE	R		C	
	DIMENSIONS (CM)	3.8, 0.43, 2.5, 0.43		3.2, 0.26	
3	ULT. LOAD (KN)	12.5	4.3	—	2.1
	SHAPE	C	C		C
	DIMENSIONS (CM)	4.4, 0.26	2.8, 0.31		2.8, 0.18
4	ULT. LOAD (KN)	36.0 ✓	—	14.4	10.3
	SHAPE	C		C	C
	DIMENSIONS (CM)	3.75, 0.51		4.7, 0.16	3.1, 0.19
5	ULT. LOAD (KN)	2 @ 12.5	—	5.2	6.9
	SHAPE	R		TENSION STRAPS	I
	DIMENSIONS (CM)	4.75, 0.[illegible]3, 2.5, 0.15			8.1, 0.15, 5.1, 0.15
6	ULT. LOAD (KN)	22.4 ✓	—	9.5	5.9
	SHAPE	C		C	C
	DIMENSIONS (CM)	3.5, 0.5		4.5, 0.09	2.0, 0.33
7	ULT. LOAD (KN)	2 @ 12.5	—	10.4	5.9
	SHAPE	R		R	I
	DIMENSIONS (CM)	4.75, 0.13, 2.5, 0.15		4.8, 0.13, 4.8, 0.13	10.5, 0.15, 5.1, 0.15
8	ULT. LOAD (KN)	2 @ 11.9 ✓	—	10.7	5.9
	SHAPE	R		R	I
	DIMENSIONS (CM)	3.5, 0.22, 2.4, 0.22		3.3, 0.48, 2.5, 0.23	8.7, 0.10, 4.6, 0.1

KEY TO DATA

SHAPE DENOTED BY C
DIMENSIONS:
5.0 x 0.125 CM



SHAPE DENOTED BY R
DIMENSIONS: 5.0, 0.5, 2.5, 0.4 CM



SHAPE DENOTED BY I
DIMENSIONS: 4.0, 0.3, 2.0, 0.4 CM

NOTES:

*SHEAR WEB FOR CONCEPT 2 —2.5-CM-DEEP ALUMINUM CCRE, 0.025 CM GRAPHITE EPOXY FACE SHEETS

**BATTEN OR LATERAL

- DATA OF TELESCOPING MEMBERS REFER TO INNER MEMBER.
- MATERIAL IS T300/934 GRAPHITE EPOXY COMPOSITE UNLESS NOTED BY ✓ (ADJACENT TO LOAD) WHICH DENOTES T75S/934.
- SIZING IS TO ADOPTED STRENGTH AND STIFFNESS REQUIREMENTS.

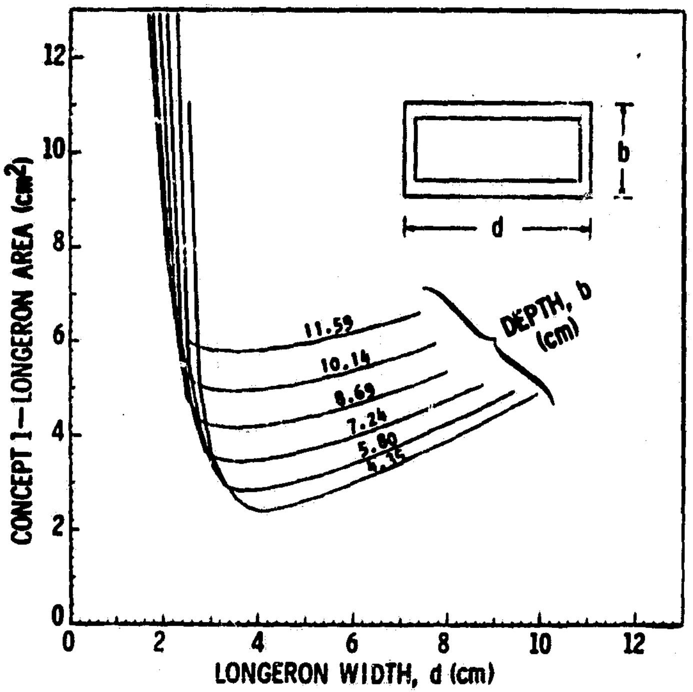


Figure 1.4-4. Example of Column Sizing Data

o For the longerons, data such as shown in Figure 1.4-4 were compared with the AE requirement compatible with the adopted flexural (EI) stiffness requirement of 2×10^8 nm^2. For example, for Concepts 1, 2, 4, 6 and 8, the AE requirement was more critical than the column stability requirement. For these cases the higher modulus material was used to maintain good packaging efficiency and/or avoid solid members. Note that for an EI requirement equal to half the adopted value, the T300/934 would be used. The cost penalty with use of the P75S/934 was accounted for in the fabrication cost analyses. For all the longerons the sizes were based upon the best compromise, through discussion with the designer, in regard to packaging, suitability of end joint attachment, and space for the folding joint.

o For the diagonal, battens, and/or pyramidal members, the data such as shown in Figure 1.4-4 were developed with sizes determined based upon the best compromise with the designer as described above. The sizes obtained were checked (excepting Concepts 2 and 5) for satisfaction of the adopted torsional stiffness (GJ). For Concepts 1, 3, 4, 6, 7, and 8 the designs satisfied the adopted torsional stiffness of 5×10^6 Nm^2 (Concept 3 in fact had a GJ five times greater than that required). For Concept 5 the X-bracing members AE values were sized to satisfy the GJ requirement. For Concept 2, NASTRAN analysis was used. The NASTRAN analysis included elastic stability analysis and shear stiffness analyses of the individual panel, including the cutouts (reinforcing around cutouts). Further,

since the total Warren truss design included an offset between the longerons and diagonal braces, a two-bay model of the structure was made.

The data of Table 1.4-1, as determined by the above discussion, are shown on the structural drawings of the eight candidate concepts. Additional discrete analyses were performed to support the concept development and mass properties data tabulated in Section 1.4.5. These analyses encompassed the housings, adapters, utilities support trays, and launch support cradles.

1.4.2 Eight Candidate Building-Block Concepts

1.4.2.1 Concept 1

The structure (Figure 1.4-5) is a pentahedral truss formed essentially from one-half of the General Dynamics tetrahedral truss, but with three differences:

- o The concept folds only in the axial direction.
- o A telescopic diagonal is provided across the base of the pyramid-shaped bay.
- o Members of various cross-sections (e.g., □ or I) are used which nest one within another, thus permitting a higher packaging ratio.

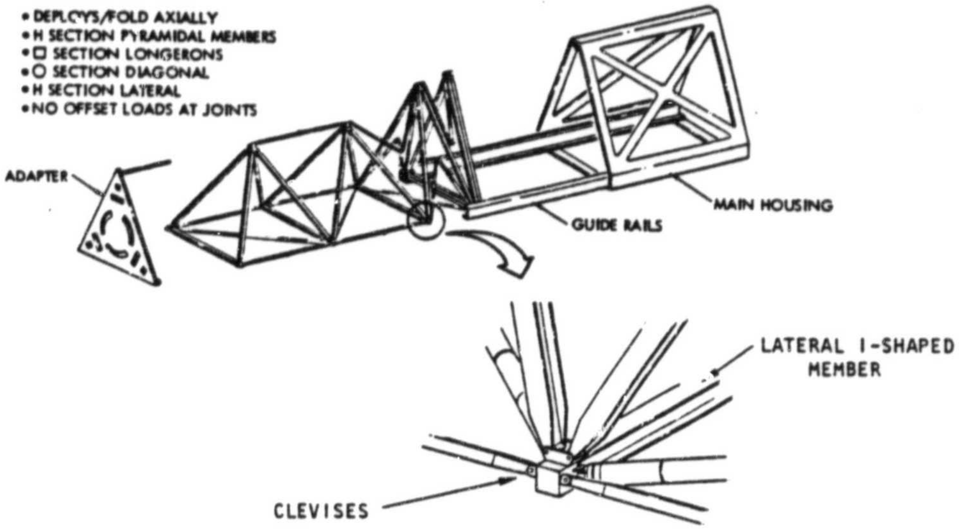


Figure 1.4-5. Concept 1

The folded truss is stowed in a rigid triangular housing from which it is deployed one bay at a time along a pair of guide rails by means of a reciprocating mechanism. To enable the truss to develop root stiffness while deploying, the rails must be a minimum length of two bays plus the stack length. The guide rails may be folded back alongside the housing for ease of stowing in the orbiter.

Subsequent to full deployment, root stiffness is developed by attachment of the truss to the rear face of the housing.

The adapter is a rigid assembly which attaches to, and moves with, the far end of the truss. It is the interface for payloads or modules and contains mechanical/structural latches, electrical/fluid interfaces and an alignment system.

The electrical utilities are mounted in trays which are attached to the truss pyramidal members and pivot about the tray centerline (Figure 1.4-6). There is ample room for the full complement of utilities. The fluid lines are installed on the squa) face of the truss in a series of pivoting trays.

The packaging ratio is 15 to 1.

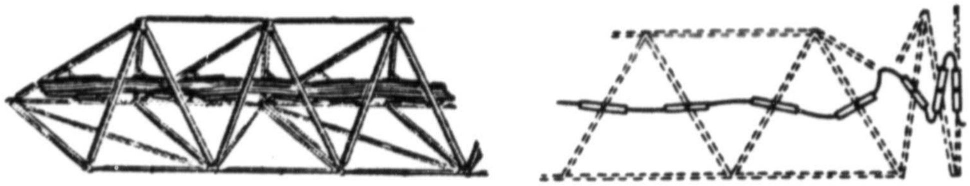


Figure 1.4-6. Installation of Utility Trays (Concept 1)

1.4.2.2 Concept 2

This concept is a deployable Warren truss, as shown in Figure 1.4-7. The truss is supported in a housing during orbiter launch and is deployed from it along a pair of rails by a reciprocating mechanism. The length of the housing is dependent on the number and size of the bays to be packaged. For the truss to be under control during deployment, the minimum length of the rails is equivalent to the length of two bays plus stack length. This will usually provide fairly long rails which need to be folded for convenient packaging in the orbiter. During deployment, the root strength of the truss is developed by the truss attachments rolling in the deployment rails. Subsequent to full deployment, the truss root strength is developed by the attachment of the truss to the housing structure. The truss consists of folding longerons, telescoping diagonals and rigid sandwich panels that form the shear panels. Because the panels are rigid, there is a choice of design such as machined waffle or honeycomb panel.

Each longeron is hinged at the middle, and the two half-longerons fold one inside the other. There is an offset between the axes of the diagonals and the longerons.

The adapter is a square, rigid assembly which also acts as the last shear panel in the truss. It is the interface for payloads and other modules and contains all of the elements necessary for alignment, berthing and utility interfacing.

The design of the shear panels is particularly suitable for the installation of utilities in trays (Figure 1.4-8). The electrical power and the signal/data cables are in two separate trays, pivoting in slots cut in the shear panels. The fluid lines which use elbow fittings, to avoid small bend radii, are mounted in similar fashion in other slots in the shear panels.

The packaging ratio is 21.6 to 1.

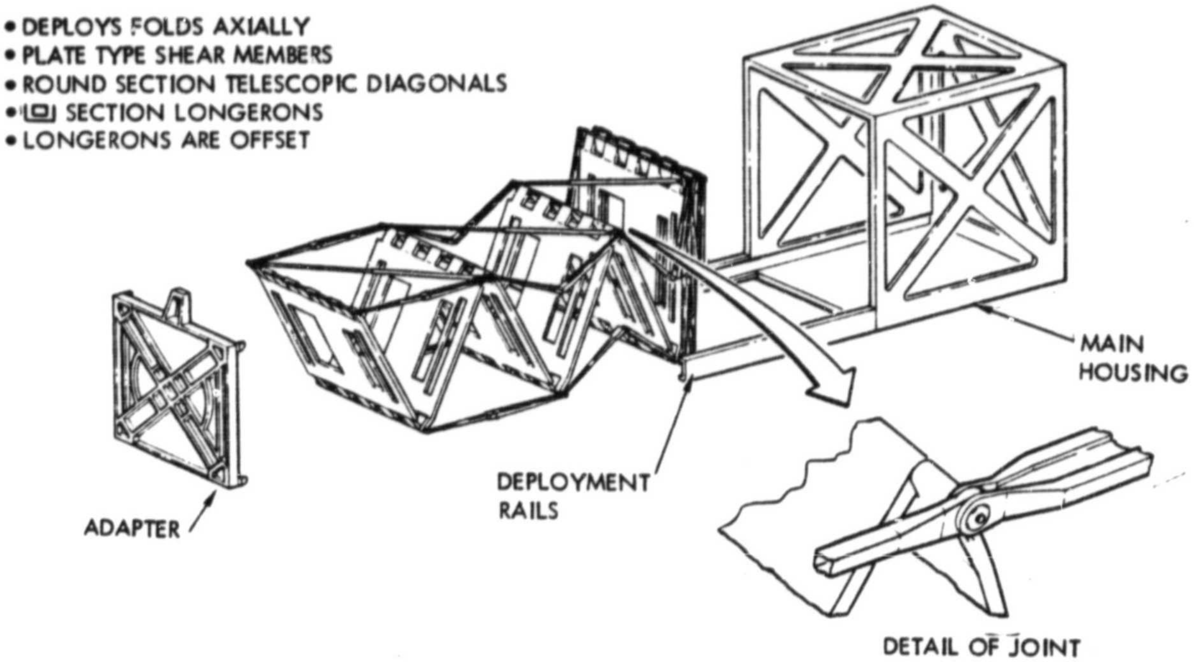


Figure 1.4-7. Concept 2

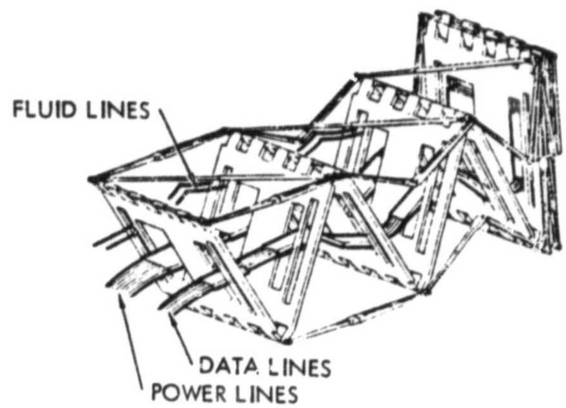


Figure 1.4-8. Installation of Utilities (Concept 2)

1.4.2.3 Concept 3

Concept 3, the tetrahedral truss, is the General Dynamics design used in its entirety (Figure 1.4-9). It is a good, practical design of a deployable truss which has been demonstrated as a working model on many occasions. The truss is a double-fold system which packages into a small volume. All of the members are round tubes which converge without offset at the joints. Springs are used at the joints to assist in the final deployment of struts and to lock them in position.

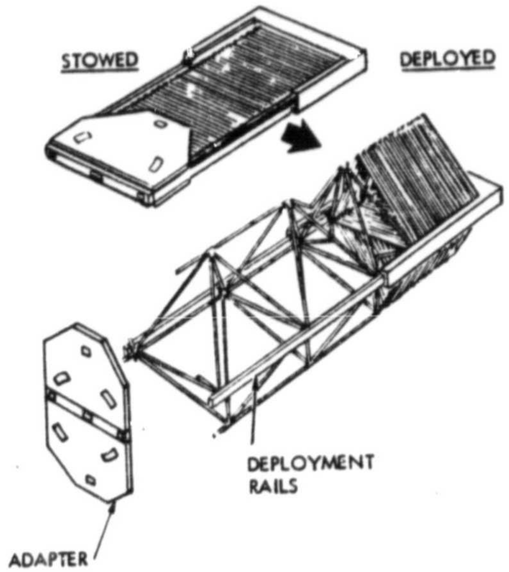


Figure 1.4-9. Concept 3

The deployment mechanism is the same as in Concepts 1 and 2, i.e., a reciprocating mechanism with folded guide rails.

During deployment the truss develops root strength through its interface with the guide rails. Subsequent to deployment, root strength is developed through the attachements between the truss longerons and the main housing, or between the truss longerons and a subsidiary structure (Section 1.4.3.3) which is deployed for that purpose.

The adapter consists of folded plates attached to the truss. It is automatically erected as the first fold of the truss is deployed. The erected adapter contains the same type of alignment, berthing and interface devices as do the rigid adapters previously described for other concepts.

The double folding of the truss causes some problems with the utilities installation (Figure 1.4-10). To install the full complement of utilities, the electrical cables are mounted on the exterior of the pyramidal members in a looped configuration. They will extend easily but must be retracted and re-looped manually. The fluid lines are mounted on the lateral members in a series of folds.

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The packaging ratio is 20 to 1 (along the length).

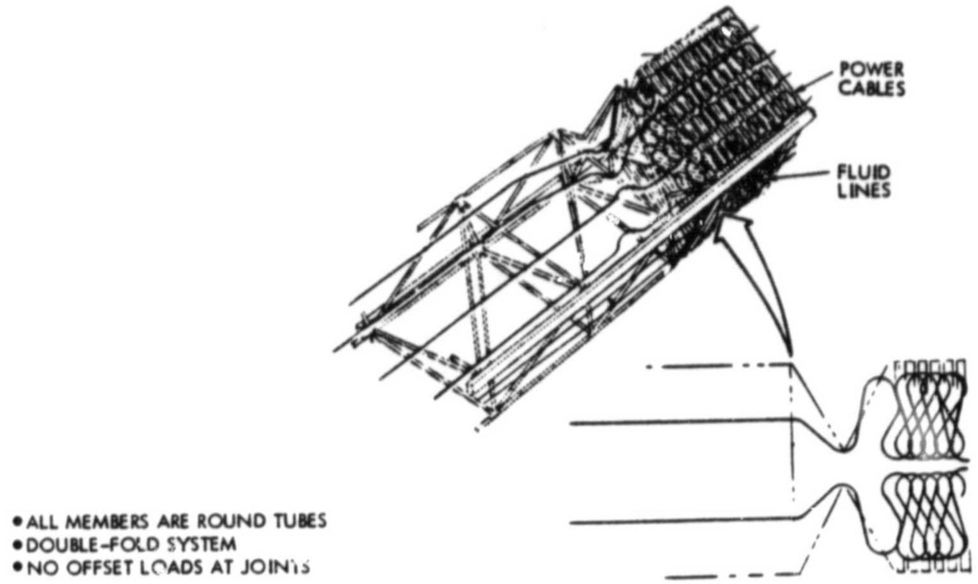


Figure 1.4-10. Installation of Utilities (Concept 3)

1.4.2.4 Concept 4

This concept (Figure 1.4-11) is a triangular truss which deploys/retracts axially without changing its end profile. Each bay consists of a rigid triangular frame, three folding longerons, and three telescopic diagonals. The three longerons folded into the stowed configuration are shown in Figure 1.4-12. It is recognized that there will have to be a support in the middle of the triangular frame to brace the longerons during launch. All of the truss members are round tubes which converge at the corner fittings without offset. The folding of the longerons toward the middle instead of in line with the batten tubes and diagonals is necessary to achieve a high packaging ratio.

The main housing is a conventional rigid framework which contains the folded truss and deployment mechanism, and carries the loads from the truss into the rest of the deployable platform. The selected deployment mechanism is a reciprocating device which deploys and rigidizes the truss one bay at a time. Initially, torsion springs were used but were replaced by a reciprocating mechanism for better packaging. During truss deployment, root strength is developed by guide rails which are stowed alongside the housing when not in use.

Because the truss does not change its triangular end profile dimensions while deploying, there is a choice of using either two or three guide rails, i.e. with two-guide rails a rail length equal to 2 bays plus the stack length is required; with three guide rails a rail length equal to 1 bay plus the stack length is required.

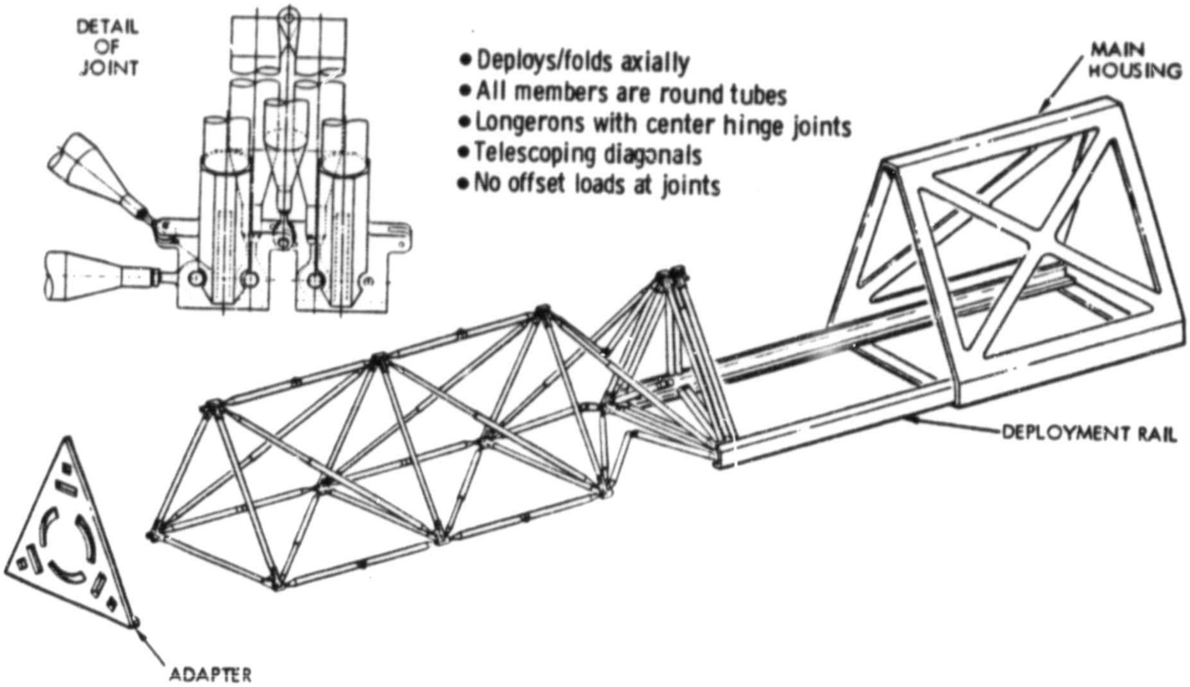


Figure 1.4-11. Concept 4

*Figure 1.4-12. End View of
Folded Configuration*

The adapter is a rigid triangular assembly which can be identical to that described for Concept No. 1.

Utilities cannot be mounted in trays because the longerons are folded into the center. Therefore, they are mounted directly onto the batten frames in a series of folds (Figure 1.4-13). For the full complement of utilities they are installed on both the inside and outside of the battens.

With the use of the reciprocating deployment system, a packaging ratio of 20.2 to 1 was achieved.

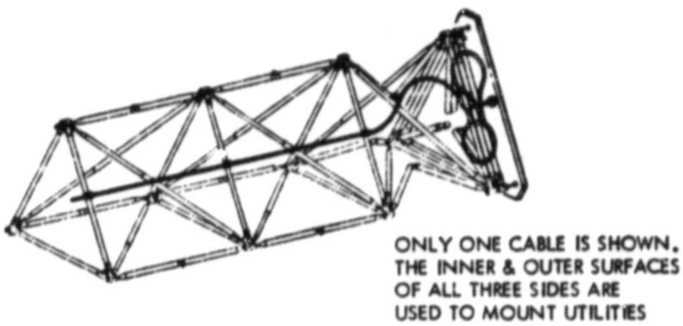


Figure 1.4-13. Utilities Installation (Concept 4)

1.4.2.5 Concepts 5 and 7

Concepts 5 and 7 are the same except that Concept 5 is a tension X-braced structure, while Concept 7 uses compression diagonals for shear and torsion capability. Concept 5 will be described (Figure 1.4-14).

Concept 5 is a triangular section truss which deploys along two axes. First, the triangular section expands to approximately 170% of its original size; then, the truss deploys along its longitudinal axis. A high ratio of deployed length/stowed length is achieved by all of the cross-bracing and longerons nesting inside the batten frames when stowed.

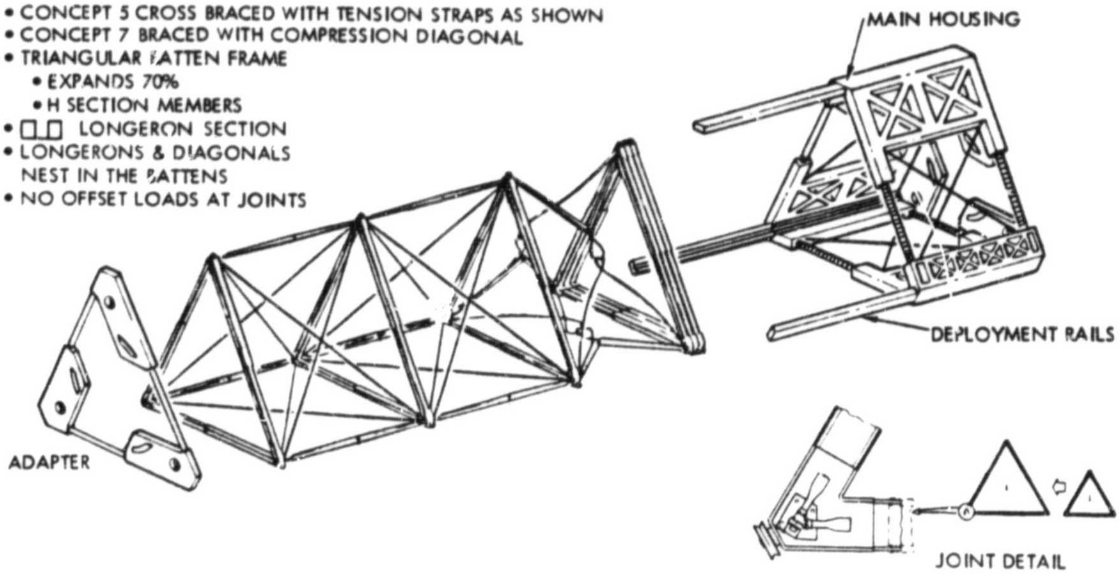


Figure 1.4-14. Concepts 5 and 7

The expanding triangular batten frame is of H-section construction. The longerons are a channel cross-section with a hinge and a latch at the mid length. The cross-braces are rectangular section tension straps, constructed of a graphite/rubber composite or similar material which has a low CTE and is flexible enough to behave as a strap instead of a rigid bar. The cross-braces are pretensioned by deploying the truss bays. The longerons serve as natural over-center tensioning devices to apply load to the cross-braces which are accurately fabricated to a predetermined length.

The expanding triangular truss is stowed in an expanding triangular housing. As the housing expands it pulls the truss with it. The housing which is made in three sections is expanded/contracted by a number of powered screw jacks. The expanding force from the housing to the truss is via the guide rails attached to the housing and the guide wheels which are attached to the truss battens. As in Concept 4 the number of guide rails can be either two or three. The longitudinal deployment mechanism is a "standard" reciprocating system using the guide rails.

Concept 7 is shown in Figure 1.4-15. Because of the double fold nature of the truss, the compression diagonal has two folding joints in addition to a telescoping joint which is considered to be a significant disadvantage.

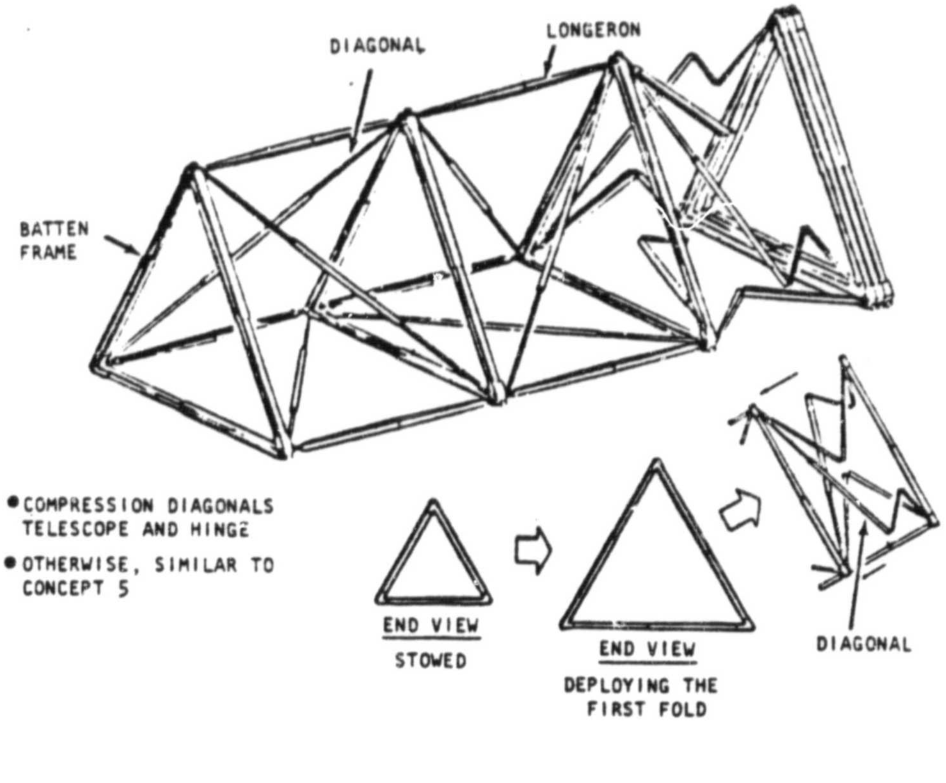


Figure 1.4-15. Concept 7

The full complement of utilities can be integrated into Concepts 5 and 7. The electrical cables are installed inside the truss triangle in trays which are pivoted at their ends from clevises mounted on the battens (Figure 1.4-16). The fluid lines are outside the truss triangle on shallow trays which are pivoted at their end from the battens.

A packaging ratio of 27.5 to 1 was achieved.

1.4.2.6 Concept 6

Concept 6 (Figure 1.4-17) is a linearly deployable truss with a square cross-section consisting of rigid batten frames joined to each other by folding longerons and telescoping diagonals pin-connected at corner fittings integral with the frames.

The longerons (Figure 1.4-18) fold toward the middle of the truss at a 45-degree angle, have self-aligning spherical ball end fittings, and hinge/lock fittings in the center of the longeron length. The telescoping diagonals have a lock mechanism and self-aligning end fittings.

X-braced tension cables can be provided for all the interior batten frames for redundancy.

All of the truss members (round tubes for the longerons, battens, and diagonals, and rectangular straps for the lateral bracing) are loaded along their centroids and converge without offsets at the batten frame corner fittings.

The stowed truss is contained in a square rigid housing to which the guide rails and deployment mechanism are attached. The number of guide rails may be 2, 3, or 4. For 2 guide rails, the rail length is equal to 2 bays plus the stack length. For 3 or 4 guide rails, the rail length is equal to 1 bay plus the stack length.

The folding guide rails and the reciprocating deployment mechanism are the same as in the concepts described previously. Initially, torsion springs were considered. The method of deployment is "one bay at a time," with root stiffness developed by the guide rails during deployment. The longerons attachment to the rear of the main housing provides the load capability subsequent to deployment.

The adapter is a square rigid assembly at the far end of the truss. It contains all of the electromechanical latches, alignment features and connections required for a payload/module interface.

The full complement of utilities is attached to the batten frames in figure-8 loops (Figure 1.4-19). Both the inside and the outside of the batten frames can be used (Figure 1.4-18).

With the use of the reciprocating mechanism, a packaging ratio of 20.4 to 1 was achieved.

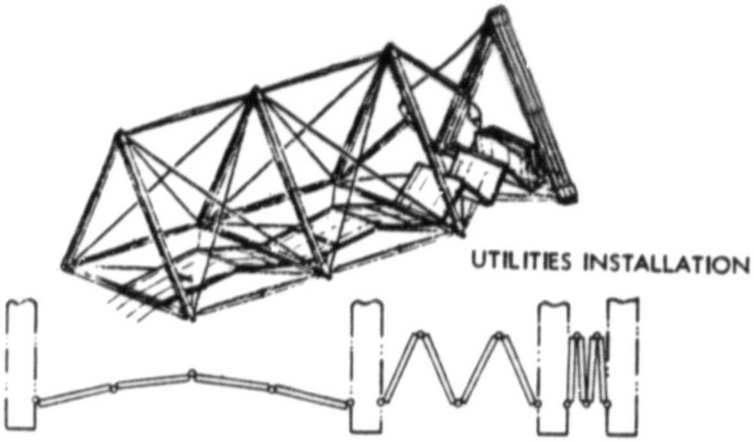


Figure 1.4-16. Utilities Installation (Concepts 5 and 7)

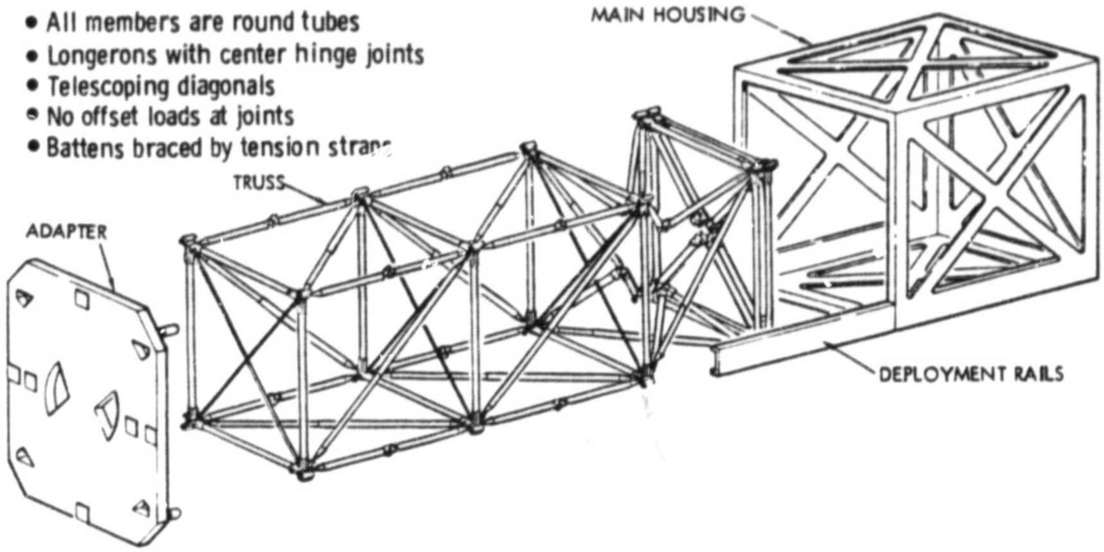


Figure 1.4-17. Concept 6

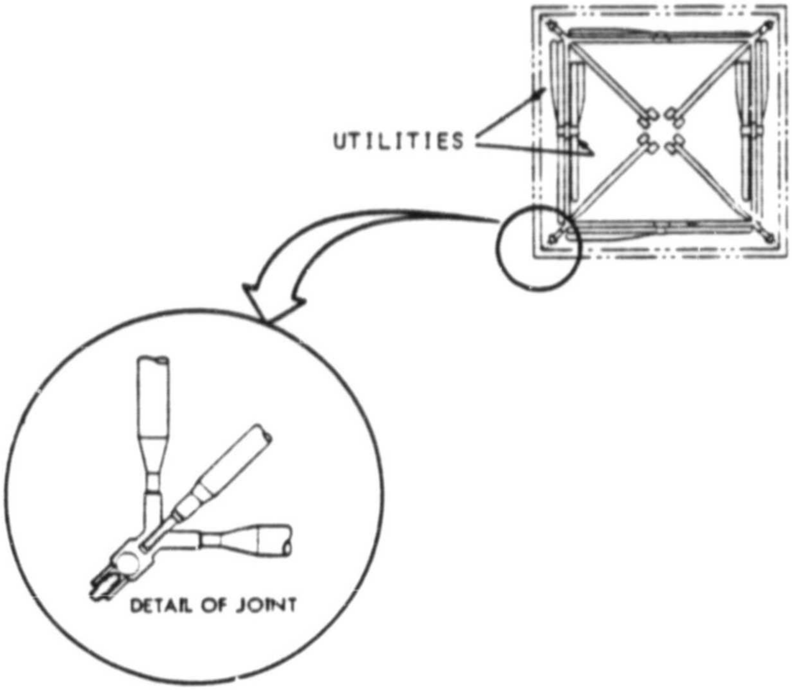
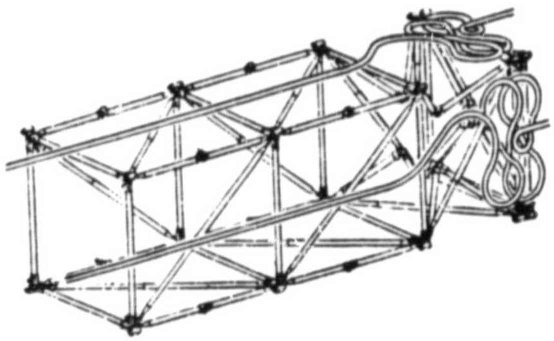


Figure 1.4-18. End View Stowed



FOR CLARITY, ONLY TWO LINES ARE SHOWN

Figure 1.4-19. Utilities Installation (Concept 6)

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1.4.2.7 Concept 8

Concept 8 (Figure 1.4-20) consists of square rigid batten frames connected together by folding longerons and telescoping diagonals. The batten frames are H-section, cross-braced by thin tension straps (as required in the interior bays for redundancy). Each longeron consists of two rectangular section members with locking hinges at their centers. The diagonals are rectangular sections which telescope within each other. The diagonals and longerons stow within the confines of the H-section battens. In the deployed configuration all the members converge at a common point.

The housing, rails, deployment system and end adapter are the same as in the square truss of Concept 6.

Concept 8 has the advantage of utility trays which can extend the whole width of the interior of the batten frames if desired, permitting a large separation between power and data lines plus the advantage of growth in the number of lines (Figure 1.4-21).

A packaging ratio of 22 to 1 was achieved.

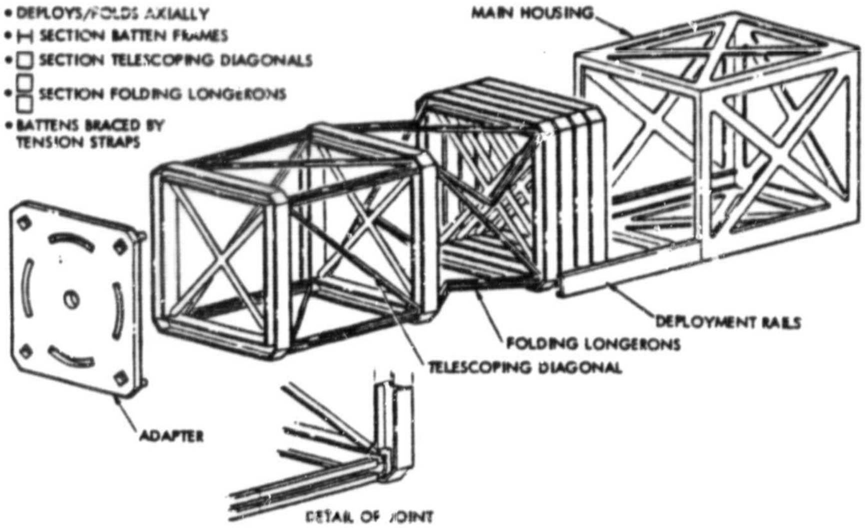

Figure 1.4-20. Concept 8

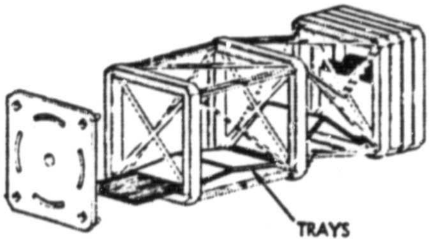

Figure 1.4-21. Utilities Installation (Concept 8)

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1.4.3 Orbiter Packaging

This section describes the studies and results applicable to the packaging of the generic platform (Section 1.2), constructed from the eight candidate building blocks, into the orbiter. The packaging is based upon the packaging efficiencies shown in the preceding section.

1.4.3.1 Concepts 1 & 4

Concepts 1 and 4 are axially folding trusses of triangular cross-section. Because of its better packaging ratio, Concept 4 uses less space (7.5 M) in the Payload Bay than does Concept 1 (9.0 M), Figure 1.4-22. Apart from this, the two concepts are identical in packaging and deployment. The figures and description apply only to the generic platform, although many other configurations can be built/deployed by following the approach outlined in this report.

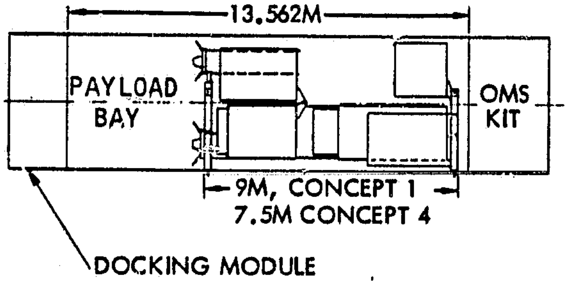

Figure 1.4-22. Orbiter Packaging —Concepts 1 and 4

The platform system deployment is 100% automatic without resort to EVA, RMS assembly or to building fixtures. All of the necessary modules, orbiter supports and cradles are incorporated in the design.

There are eight triangular building blocks and two cradles. The cradles are structures which interface with the orbiter trunnion and keel fittings, join some of the building blocks together at their hinge points, and provide mounting surfaces for the spacecraft systems.

Each of the building blocks is a single truss design, i.e., only one truss is deployed from each housing.

The platform is assembled and checked out on the ground before it is folded and placed in the orbiter. All of the utility connections from end to end of the platform are made on the ground and are not broken during stowage or deployment.

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The stowed platform is removed from the orbiter by using the RMS, and remains in the grasp of the RMS and/or is attached to the HAPA during the initial stages of deployment. If continuous operation and a low deployment rate of 3/cm per second are assumed, the platform could be deployed in about 72 minutes. Some trusses deploy simultaneously. There are three basic operations involved in deploying the platforms, i.e., reorienting the building blocks, extending the trusses, and latching.

One sequence of operations to deploy the platform is shown in Figure 1.4-23. Several other sequences are possible in achieving the same configuration. The solar array is not shown as an integrated deployable item in this design, although it is quite possible that it could be incorporated. Items such as large payloads, modules, etc., are added to the deployed platform by the RMS, using the interfaces provided.

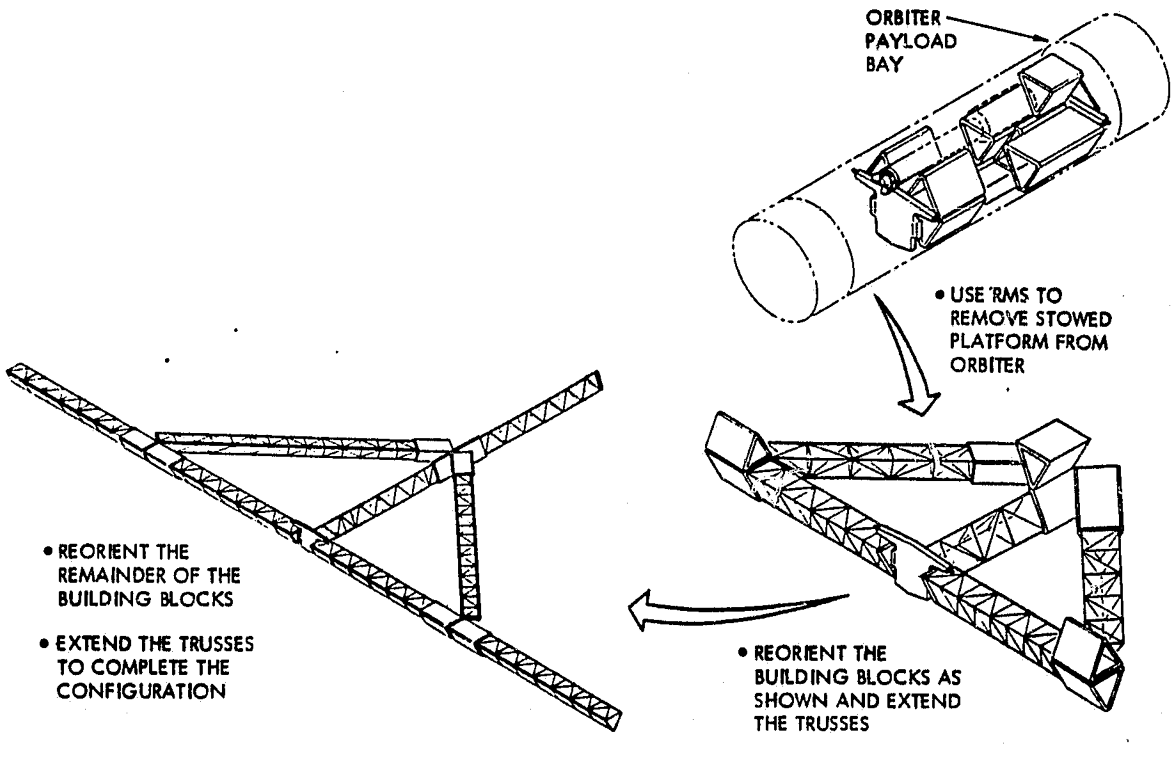

Figure 1.4-23. Platform Deployment—Concepts 1 and 4

1.4.3.2 Concepts 2, 6 and 8

This section discusses the packaging and deployment sequence for axially deploying building blocks, of square cross-section of which Concepts 2, 6, and 8 are typical examples. The three concepts are identical in packaging and deployment with the exception of the slight difference of their lengths in the payload bay (Figure 1.4-24.).

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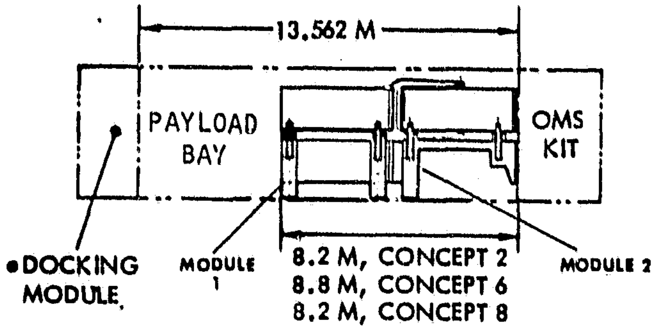

Figure 1.4-24. Orbiter Packaging —Concepts 2, 6, and 8

There are five building blocks and two modules in the assembly (Figure/1.4-25). Of the five building blocks two are "singles", i.e., one truss from a main housing, and three are "doubles", i.e., two trusses, one from each end of a main housing.

- o Module 1 mounts three of the building blocks and is a cradle which supports them in the orbiter. It forms part of the deployed platform and is used to mount spacecraft systems and equipment.
- o Module 2 mounts the remaining two building blocks and is the cradle which supports them in the orbiter. It too is deployed with the rest of the platform and is used to mount spacecraft systems.

The complete package is removed as a unit from the payload bay using the RMS and is retained on the RMS (or mounted onto the HAPA) for the initial stages of deployment. The stages of the deployment sequence are:

- o Reorient four of the building blocks by rotating them until all five building blocks are in the same plane.
- o Extend the trusses as shown to form the vertical of the T shape of the platform.
- o Rotate the building blocks which form the diagonals and extend the trusses to complete the T shape of the platform.

The platform is automatically deloyable without the assistance of EVA, building/assembly fixtures or piece-by-piece assembly using the RMS. There is no requirement to make electrical or fluid connections in flight except where separate units such as payloads are added subsequent to deployment by the RMS.

1.4.3.3 Concept 3

Although this concept was successfully packaged (Figure 1.4-26) and deployed, this study does point up the difficulties with the use of a double-fold structure. The advantages gained from the dense volumetric packaging of the structure are largely illusory when one turns from a simple truss to a complete deployable platform system. Some of the problems which contribute are:

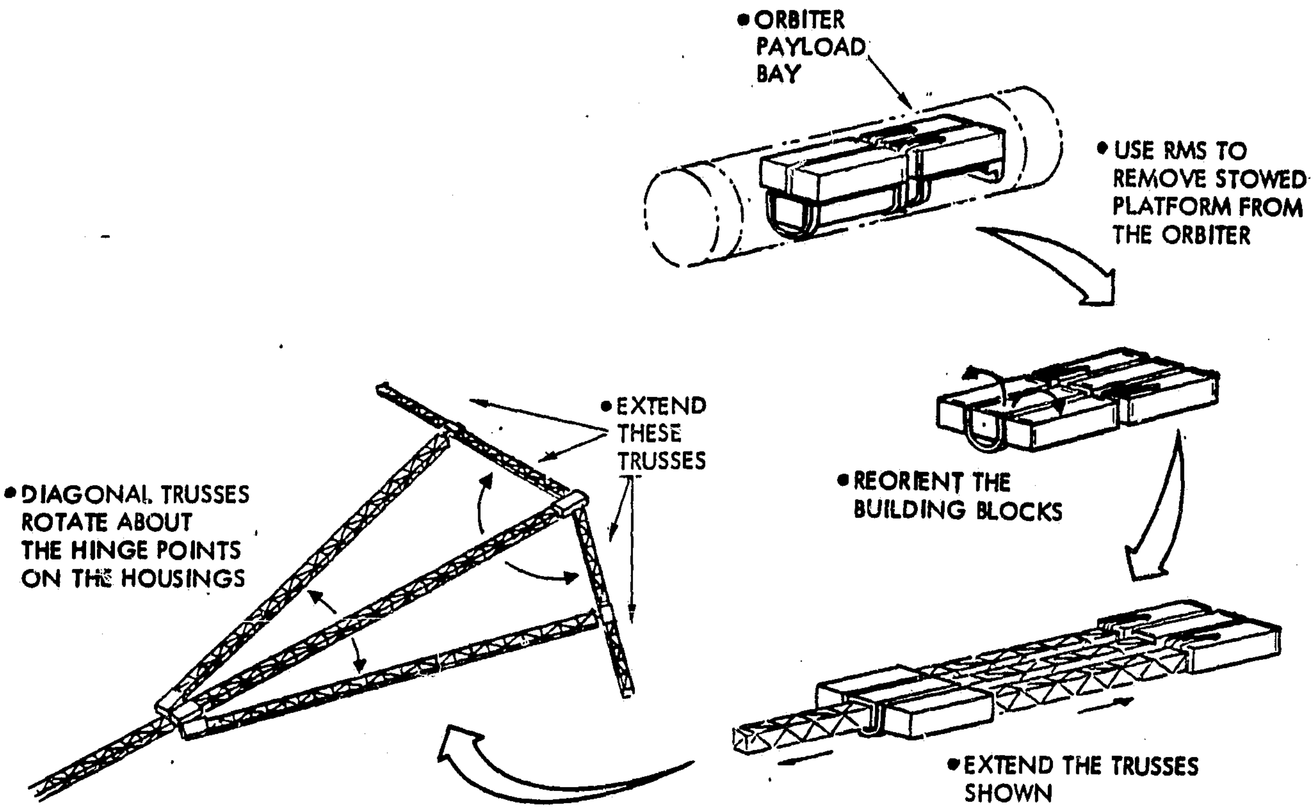

Figure 1.4-25. Platform Deployment—Concepts 2, 6, and 8

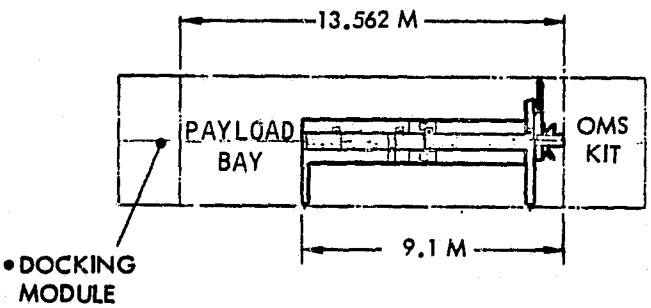

*Figure 1.4-26. Orbiter Packaging
—Concept 3*

- The stowed double-fold truss is difficult to support in the orbiter with provision of stiffness compatible with frequency separation (10 Hz) from the orbiter.
- The methods of joining truss to truss, or truss to module, are more complex. It is sometimes necessary to erect a subsidiary structure for structural continuity.
- The end adapter cannot be a simple plate to support latches, interfaces, etc.
- Mounting of the adopted complement of utilities is difficult because of limited space.

In spite of these difficulties, the generic platform can be built using Concept 3. It is an automatic deployable platform which requires no EVA, no building/assembly fixture and no part-by-part erection. The electric/fluid utilities are installed and checked out on the ground from end to end of the platform. There are no connections broken/reconnected between stowage into the orbiter and final deployment.

The platform consists of five building blocks and one module. Of the five building blocks, three are "double building blocks," i.e., a truss deploys out of each end of the housing. The module is an integral part of the platform, is deployed along with the building blocks, is intended to mount spacecraft systems, join three of the truss together, and serves as a base for deploying subsidiary structure for reacting truss loads.

The packaged platform (Figure 1.4-27) is removed from the orbiter by means of the RMS. While it is in the grasp of the RMS (or HAPA) the initial stages of platform deployment are made. There are several stages in the deployment sequence.

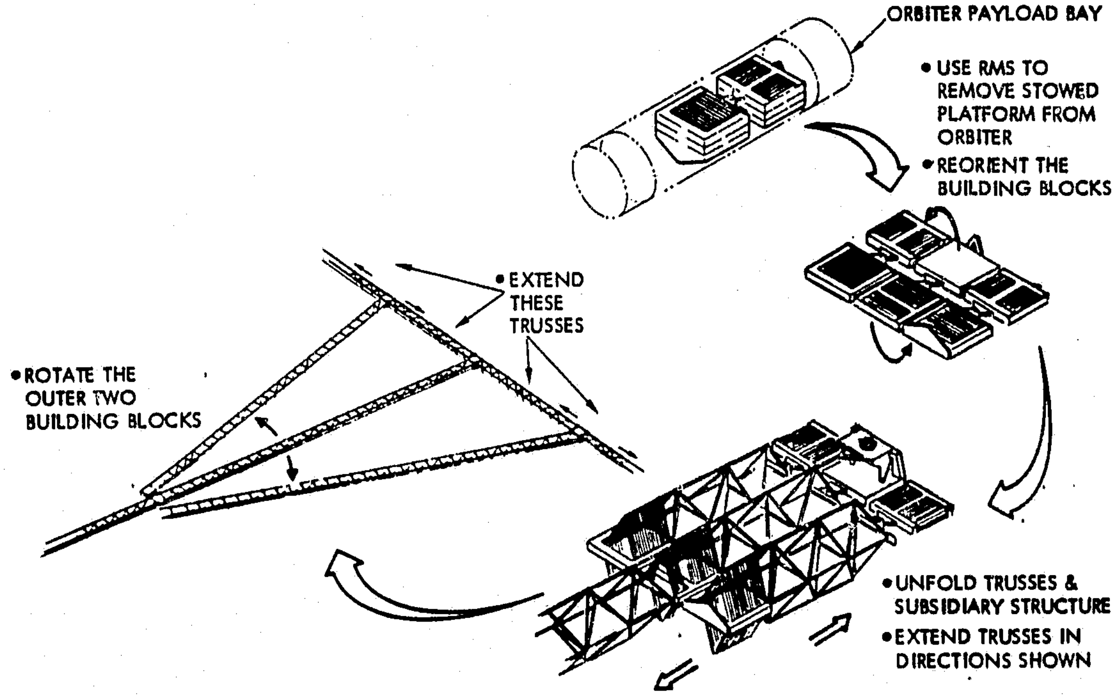

Figure 1.4-27. Platform Deployment—Concept 3

- o Unfold the building blocks.
- o Deploy the subsidiary structure on the module. This includes a docking port and attach points for reacting truss loads.
- o Erect the first fold of the trusses.
- o Extend the trusses part way.
- o Make the joints at the module attach points.
- o Extend the trusses completely.

This completes the deployment of the platform. Payloads/solar array/RCS pods may be added, as required, by a second launch or by using the empty portions of the payload bay of the first launch.

1.4.3.4 Concepts 5 and 7

Concepts 5 and 7 are stowed in the orbiter (Figure 1.4-28) and deploy to the final configuration in identical fashion. The stowage and deployment shown are for the generic platform only. It is a completely automatic deployable system which requires no EVA and no assembly or erection by RMS. The deployable platform includes six buiding blocks and two modules.

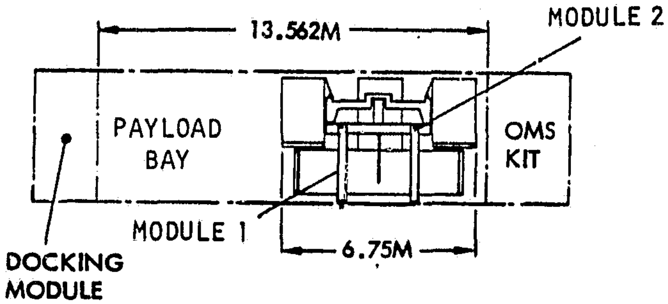

Figure 1.4-28. Orbiter Packaging —Concepts 5 and 7

- o Module 1 mounts three of the building blocks and is a cradle which supports the whole of the stowed platform in the orbiter. It forms part of the deployed platform and is used to contain spacecraft systems.
- o Module 2 mounts the remaining three building blocks. It does not interface directly with the orbiter, but is attached to Module 1. It too forms part of the deployed platform and is used to mount spacecraft systems.

The complete package is removed from the orbiter in its stowed form by using the RMS. While it is in the grasp of RMS (or HAPA) automatic deployment is initiated. There is no requirement for a building/assembly fixture to hold the platform while the RMS attaches trusses or modules section by section. As in all of the Rockwell platform deployment concepts, the utilities are installed and checked out on the gound and not disturbed thereafter.

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Because of the high packaging efficiency, it is possible to stow some of the building blocks with their lengths normal to the major axis of the payload bay, i.e., stow them "across" the ship instead of lengthwise. This permits arrangements such that the building-blocks do not have to be reoriented to achieve the deployed configuration.

There are only two stages in the deployment sequence (Figure 1.4-29), i.e., expand the triangle shapes of the building blocks and extend the trusses.

Of all the concepts which were studied, Concepts 5 and 7 proved to be the most suitable for packaging and deploying the Generic Platform. They stowed in the shortest length of the payload bay and required the least reorientation of the building blocks.

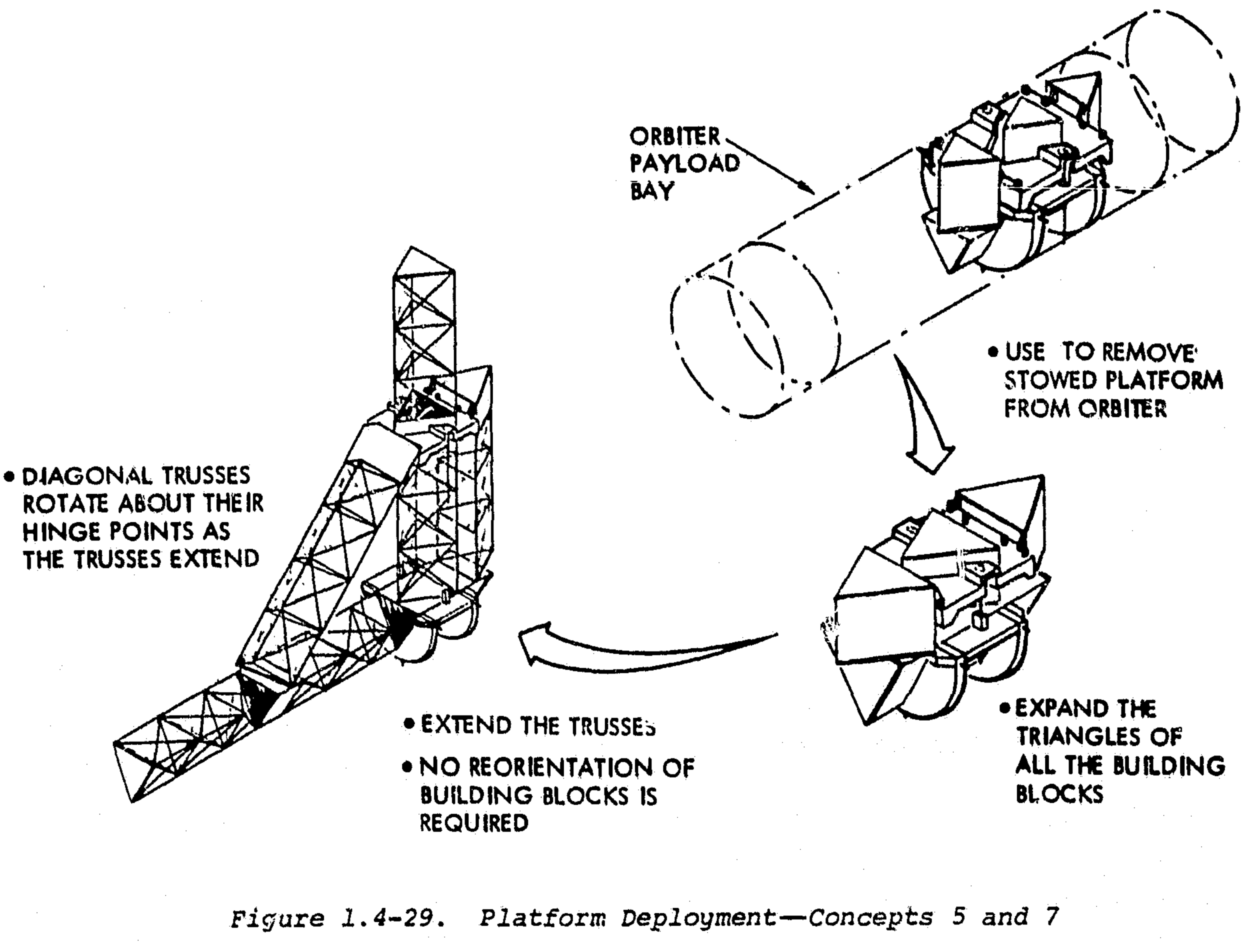

Figure 1.4-29. Platform Deployment—Concepts 5 and 7

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1.4.4 Thermal Analysis

A thermal analysis provided a prediction of the thermal gradients across the candidate deployable trusses (Table 1.4-2). The temperatures shown are the average over the length of each member. These gradients were determined for the structural arrangement/geometry shown in Figure 1.4-30 and for GEO applications. LEO applications have lower gradients due to earth and albedo heating.

Table 1.4-2. Thermal Gradients between Longerons

CONCEPT	d (m)	DIAGONAL PRESENT	LONGERON TEMPERATURE (°C)		ΔT (°C)	FIGURE OF MERIT d/(ΔT)
			SUN SIDE	SHADE SIDE		
1	1.6	YES	21.6	-13.5	35.1	0.046
2	1.3	YES	20.7	-11.6	32.3	0.040
		NO	24.0	4.8	19.2	0.067
3	3.0	NO	24.0	2.7	21.3	0.140
4	1.7	YES	21.7	-7.4	29.1	0.058
5	2.5	YES (TS)*	23.8	-36.0	59.8	0.041
6	1.3	YES	22.5	-22.6	45.1	0.029
7	2.5	YES	23.6	-56.7	80.3	0.031
8	1.3	YES	22.5	-17.5	40.0	0.031

*(TS) = TENSION STRAP

The thermal gradients are due to shadowing when one or more structural members pass between another member and the sun during orbit.

For analysis, each configuration can be represented as a Z-section (Figure 1.4-31) or as two parallel members if a diagonal is not present in the plane. A thermal math model of the Z-section is used for the analysis. It consists of 24 nodes and provides for conduction between nodes. The construction material is graphite composite. The surface radiation properties were assumed to be nearly black. Emmittance and absorptivity values of 0.85 were used. The thermal model for the two parallel member cases is a single node heat balance.

Two important assumptions in the analysis are the estimate of the shadow time and the view factor for solar impingement during shadow. The shadow time t_s (minutes) is estimated by the following expression. For two parallel members,

$$t_s = 8(0.2566 + \sin^{-1} D/S)$$

where D = member diameter
S = distance between members

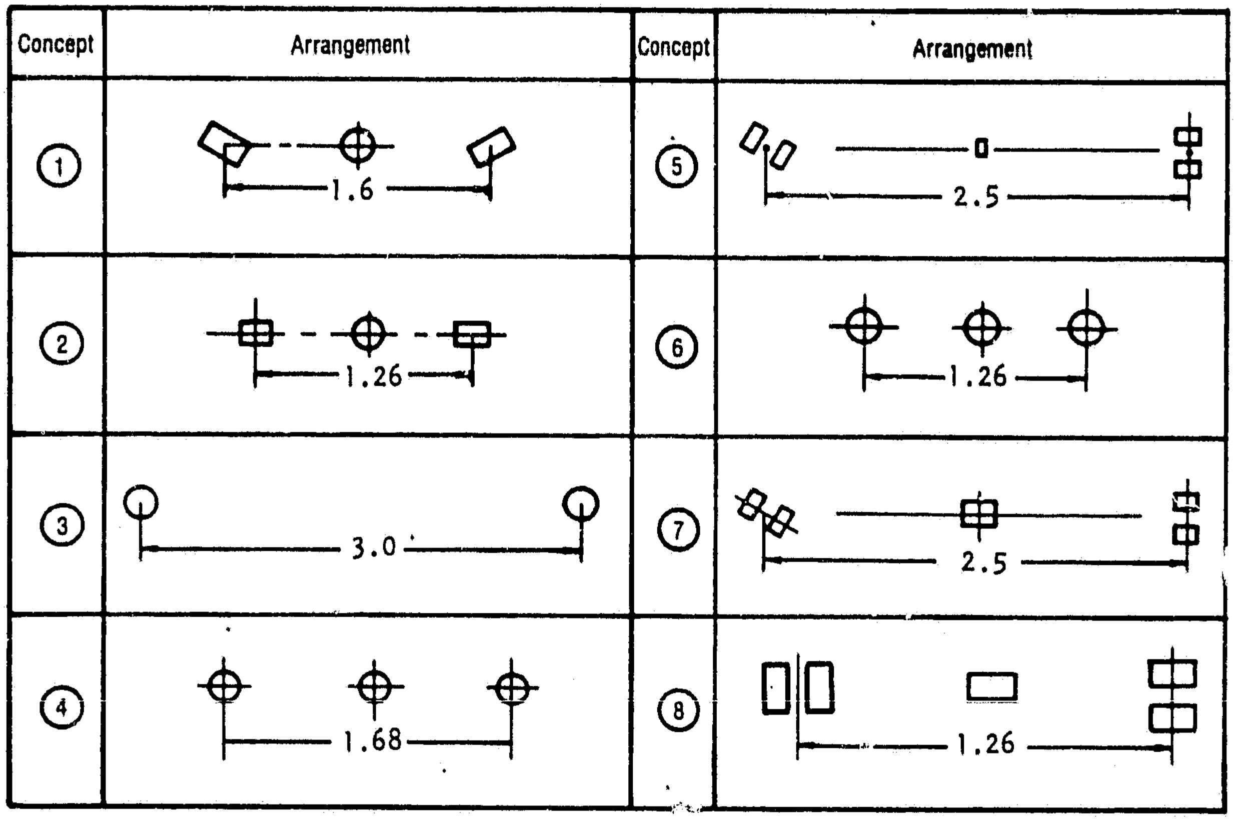

Figure 1.4-30. Structure Arrangement for Thermal Gradient Analysis (Dimensions in Meters)

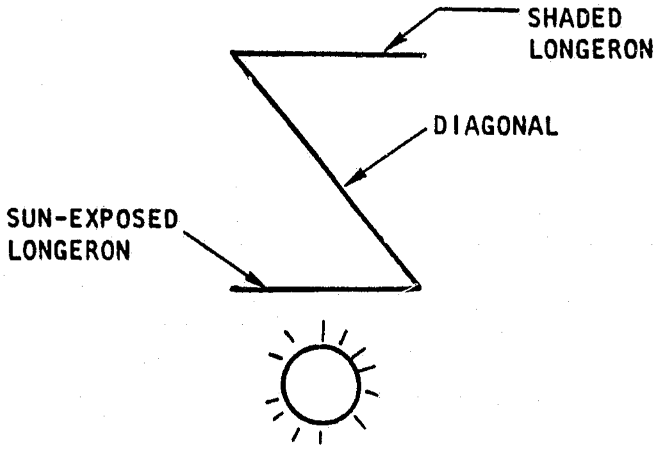

Figure 1.4-31. Thermal Analysis Configuration

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This expression accounts for the size of the shadowing member, the distance between members, and the included angle of the sun. For the Z-section, a modified form of this expression was used. The distance between members was allowed to vary with nodal position.

The view factor for solar impingement of the shadowed member was approximated by the following cosine function,

$$F = \cos^2 (180\ t/t_s)$$

This function has a value of unity at the extremes and a value of zero at the center of the time interval.

Table 1.4-2 also illustrates the parameters to evaluate the relative merit of each concept for thermal stability. The end rotation of a beam θ is determined by

$$\theta = \frac{\alpha(\Delta T)\ell}{d}$$

For beams of equal length and the same material, then θ is proportional to $(\Delta T/d)$ or the figure of merit is proportional to $(d/\Delta T)$ with high values being best. The foregoing data were based upon the structural design sizes as determined by the strength/stiffness characteristics, packaging, minimum weight, and joint design considerations. The design was not adjusted for reduction of the thermal gradients. For example, for Concept 6 the thermal gradient can be reduced by reduction of the diagonal diameter (reduced shadow) with a small weight impact (of no consequence to LEO platforms). Also, since the data are based upon $\alpha/\varepsilon = 1.0$ with ε and $\alpha = 0.85$, if reduction of the thermal gradients is required, initial values of $\alpha = 0.10$ can be realized by wrapping the graphite composite structure with silver teflon tape. Though α will increase over the life of the structure, particularly in the GEO environment, the end of life value will be less than 0.85.

A perspective on these thermal gradients is appropriate. For a 40-meter cantilever using concept 6, the end will have a thermal induced rotation of 0.028° ($\alpha = 0.36 \times 10^{-6}$ m/mC) due to the specified gradient of 45°C across a design with a 1.26 m depth. (The 40 m length is used since precision antennas would be mounted closest to the center of the configuration). Section 1.1-10 indicates the desired pointing accuracies should be 0.05 to 0.10 degree (based on statistical analyses). If necessary, a value of 0.28° can be reduced by many of the techniques delineated above. In addition, placement of the square cross-section as shown in Figure 1.4-32 can reduce the 0.028° rotation to approximately 0.020°. If necessary, larger trusses (Figure 6) can further reduce the end rotation. The use of composites with negative coefficients of expansion and metal fittings can also reduce these rotations.

In the light of the foregoing, it is also appropriate to note that a 22.2 N shear on the end of a 40 m cantiliver will induce an end rotation of 0.005° for the adopted flexural stiffness, or for 4.45 N induce an end rotation of 0.001°. Hence, RCS induced distortions are small by comparison.

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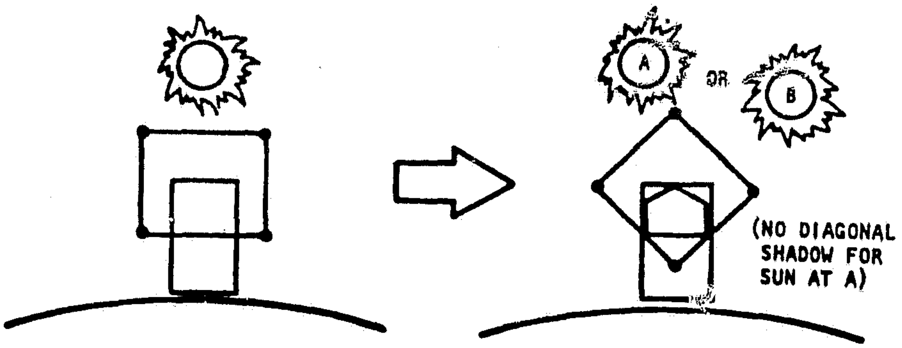

Figure 1.4-32. Approach to Reduce Thermal-Induced Truss Rotation

1.4.5 Mass Properties Analysis

The results of the deployable platform systems mass properties analysis of the generic platform contructed from Concepts 1 through 7 are shown in Table 1.4-3. These masses are determined by a combination of detailed calculations where specific structure sizing was available (deployable trusses, for example), and standard preliminary design mass estimation techniques (housings, for example). The generic platform mass is based upon the adopted strength, stiffness, and complement of utilities delineated in Section 1.1.

Table 1.4-4 illustrates the major differences in the masses for Concepts 1 through 8. The data for Concept 8 are estimated from the data provided for Concept 6 in Table 1.4-3. The data shown are used in the orbit transfer cost data described in Section 4. The data in Table 1.4-4 were also used as a basepoint for extrapolation of the weight differences for a generic platform designed to 1/10 the adopted strength/stiffness requirement (Table 4.3-5).

A review of Table 1.4-3 illustrates all of the candidate deployable platform systems, including a 20% growth allowance, are launchable to a 210 nmi orbit 28.5° inclination (Figure 1.4-33) without use of an OMS kit. The balance of mass between the values shown and an allowable 20,000 kg is available for payloads, RCS propulsion modules, etc. An addditional 2500 kg is launchable if an OMS kit is provided. The 210 Nmi orbit is considered suitable, since for the generic platform, decay to 150 nautical miles would not occur until 60 days after insertion into orbit. This is considered ample time for deployment, installation of payloads, checkout, etc.

1.4.6 Cost Analysis

Consideration was devoted, in support of the Concepts Selection Trade, to the establishment of the relative costs associated with the design, development, testing, special technology needs, fabrication, shuttle launch, and orbit transfer of GEO platforms. The design, development, and testing costs were not included since they are dependent on the number, types of application and

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Table 1.4-3. *Candidate Concepts Generic Platform Mass Summary*

ITEM	MASS (KG)						
	CONCEPT 1	CONCEPT 2	CONCEPT 3	CONCEPT 4	CONCEPT 5	CONCEPT 6	CONCEPT 7
1. BASIC TRUSS STRUCTURE	1,924	3,193	1,946	1,802	1,616	1,810	2,192
2. BASIC TRUSS JOINTS	1,325	890	965	1,652	623	1,890	1,264
3. UTILITIES INSTALLATION EQUIPMENT	813	202	118	283	749	199	749
4. UTILITIES	2,554	2,569	2,544	2,559	2,557	2,554	2,557
5. HOUSINGS	675	540	354	599	597	559	597
6. ADAPTERS	52	65	95	60	89	73	89
7. MECHANISMS	476	377	803	572	549	789	548
8. HOUSING-TO-HOUSING ATTACH. MECH.	127	156	178	128	98	156	98
9. SEPARATE DEPLOYABLE STRUCTURES	-	-	68	-	102	-	102
10. DOCKING PORTS	409	409	409	409	409	409	409
11. CONTROL MODULE EQUIPMENT	2,004	2,004	2,004	2,004	2,004	2,004	2,004
12. MISCELLANEOUS	23	91	91	23	29	91	29
SUBTOTAL	10,382	10,496	9,575	10,091	9,422	10,534	10,538
13. ORBITER INTEGRATION WEIGHT	2,240	2,547	3,440	2,154	1,130	2,774	1,130
SUBTOTAL	12,622	13,043	13,015	12,245	10,552	13,308	11,768
14. GROWTH (20%)	2,525	2,608	2,604	2,445	2,110	2,662	2,353
TOTAL	15,147	15,651	15,619	14,669	12,662	15,979	14,121

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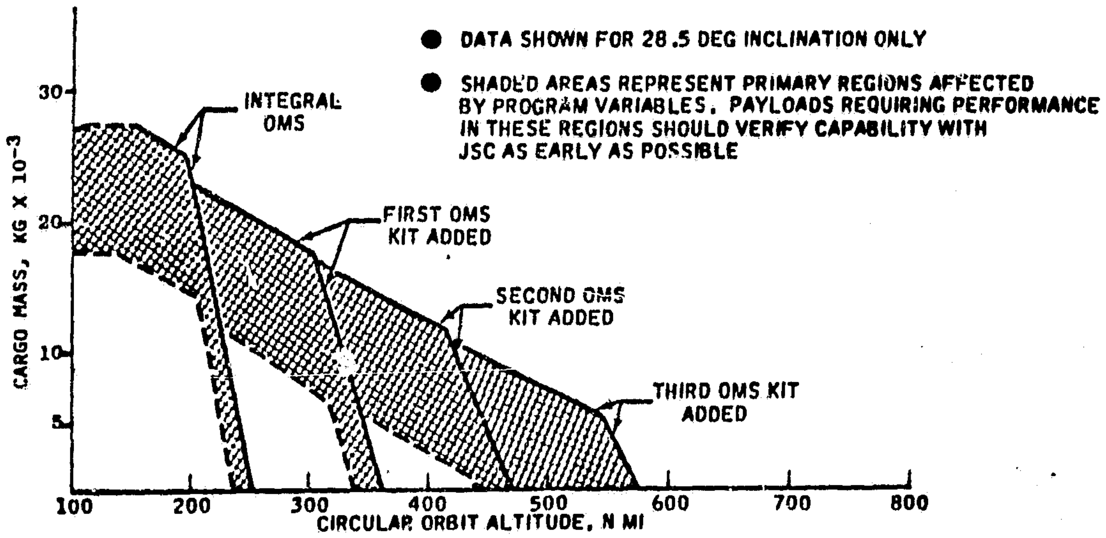

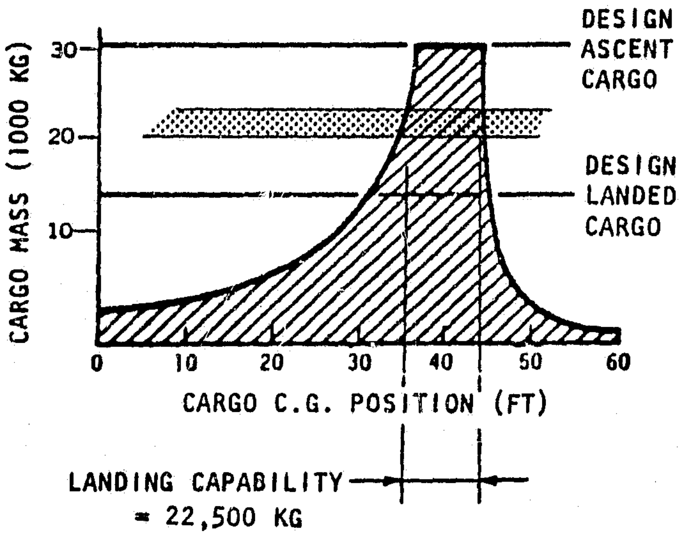

Figure 1.4-33. Orbiter Payload Mass—Launch/Landing Constraints

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Table 1.4-4. Comparative Mass ($kg \times 10^{-3}$)—Generic Platform

ITEM	CONCEPT 1	2	3	4	5	6	7	8
BASIC TRUSS ELEMENTS	1.9	3.2	2.0	1.8	1.6	1.8	2.2	1.8
ALL JOINTS	1.3	0.9	1.0	1.6	0.6	1.9	1.3	1.9
UTILITIES INSTALLATION SYSTEM	0.8	0.2	0.1	0.3	0.7	0.2	0.7	0.7
HOUSING AND ADAPTERS, HOUSING-TO-HOUSING ATTACH.	0.9	0.9	0.6	0.8	0.8	0.9	0.8	0.6
DEPLOYABLE STRUCTURES FOR DOCKING PORTS	-	-	0.1	-	0.1	-	0.1	-
TOTAL	4.9	5.2	3.8	4.5	3.8	4.8	5.1	5.0
Δ MASS	1.1	1.4	0.0	0.7	0.0	1.0	1.3	1.2

ALL DESIGNS, INCLUDING DEPLOYMENT MECHANISMS, UTILITIES LINES, DOCKING PORTS, CONTROL MODULE EQUIPMENT, AND CRADLE WEIGHT (WITH 20% GROWTH)—LESS THAN 16,600 KG.

requirements of future platforms which are not defined. The special technology development costs (as subsequently shown in Section 3) are primarily the same across all the candidate concepts, with the few exceptions resulting in negligible cost impacts (particularly if spread across several platforms).

The analysis, therefore, determined the recurring fabrication cost, Shuttle launch and for GEO platforms, the cost of orbit transfer. All the cost analysis data are expressed in terms of FY 1981 dollars and are relative, i.e., applicable only to items which are different. The costs of items that are the same across all concepts are not included. For example, the cost of the basic power, data and fluid utilities themselves, docking ports, etc., is not included.

1.4.6.1 Fabrication Cost Data

The component fabrication costs to construct the generic platform are shown in Table 1.4-5. These data are incorporated directly into Table 4-4-1. These data were derived by a combination of material and fixture costs (as appropriate) and manufacturing hours to which a composite cost rate was applied. The composite cost rate includes direct labor, overhead, and general and administrative (G&A).

The materials, fixture, and manufacturing hours data presented were determined from estimates by advanced manufacturing personnel using drawing details. Where detailed information was not available, in-house Rockwell parametric cost model techniques were used.

Table 1.4-5. Fabrication Cost Data ($ Million)

COST ITEM	CONCEPT 1	2	3	4	5	6	7
BASIC TRUSS MEMBERS (COMPOSITE)	2.2	5.0	1.0	1.7	2.0	1.9	2.3
TRUSS JOINTS AND FITTINGS	2.2	1.7	1.5	2.1	1.2	3.0	1.9
TRUSS ASSEMBLY AND CHECKOUT	0.4	0.6	0.3	0.5	0.3	0.6	0.6
UTILITIES SUPPORT SYSTEM (FABRICATION AND INSTALLATION)	0.4	0.2	0.1	0.1	0.8	0.1	0.8
INSTALLATION OF UTILITIES AND CHECKOUT	0.1	0.1	0.1	0.1	0.1	0.1	0.1
END ADAPTERS AND MAIN HOUSINGS	0.4	0.3	0.3	0.3	0.4	0.3	0.4
BUILDING BLOCK TO BUILDING BLOCK ATTACH MECHANISMS	2.3	2.9	3.5	2.2	1.7	2.8	1.7
TOTAL ($M)	8.0	10.8	6.8	7.0	6.5	8.8	7.8
Δ ($M)	1.5	4.3	0.3	0.5	0	2.3	1.3

NOTE: CONCEPT 8 ESTIMATED FROM EXISTING DATA FOR CONCEPT 6

It is pertinent to note that Table 1.4-5 does not include the costs of deployment mechanisms since the building blocks can utilize either a reciprocating device (GD design or equivalent) or a stored strain energy system. The differences between the same type mechanism across the single folded designs were regarded as negligible. In retrospect, the additional cost of the mechanisms for the double folding of Concepts 3, 5, and 7 should be included into the last cost item in the table. To do so would only enhance the conclusions and decisions made in Section 4. and require extensive. numerical change throughout the tables for the sake of consistency. Hence, this item is not included.

1.4.6.2 Shuttle Launch Cost

The Shuttle launch cost data for use in Table 4.4-1 are based upon a cost of $2.6M/meter of Shuttle cargo bay length. It is derived from an FY 1982 total cost of $48M for 18.3 m of bay length. The $48M is the FY 1981 cost extrapolated from the baseline FY 1975 launch cost of $32M and is for a medium traffic model (40 launches per year). A significant reservation on the use of this total value is that for a dedicated mission the system cost is incurred regardless of the usage of the bay. This reservation is reflected in the allocation of points in the totaling of the major criteria.

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1.4.6.3 GEO Orbit Transfer Cost

The Rockwell space station studies have identified a cost (in FY 1981 dollars) of $8,800 per kg of mass delivered to GEO orbit including the launch to LEO by the shuttle. This cost is also based on a medium mission model.

1.4.7 Miscellaneous Design Issues

During the concept development discussed throughout this section , three design concerns surfaced as follows:

- Extension of a truss which has a large payload while maintaining the deployable truss root strength with the guide rails
- Implication of building block to building-block structural attachments that are not fully fixed about all three axes
- Potential significance of "joint slop"

The first issue has been resolved as discussed in Section 1.3.3.5. The second issue has been investigated and resolved as follows:

A configuration such as the generic or ASASP platform can have joints without full structural continuity at places, such as joints 1 and 2 of Figure 1.4-34, with acceptable reductions in effective stiffness.

A sensitivity analysis of platform overall stiffness variations associated with different end joint design characteristics was performed. The analysis was performed with a NASTRAN model that simulates the ASASP platform configuration and mass distribution. The construction platform was not included. The ASASP was used since the mass distribution was more readily available and the generic platform was fashioned after the ASASP. The adopted stiffness values were used.

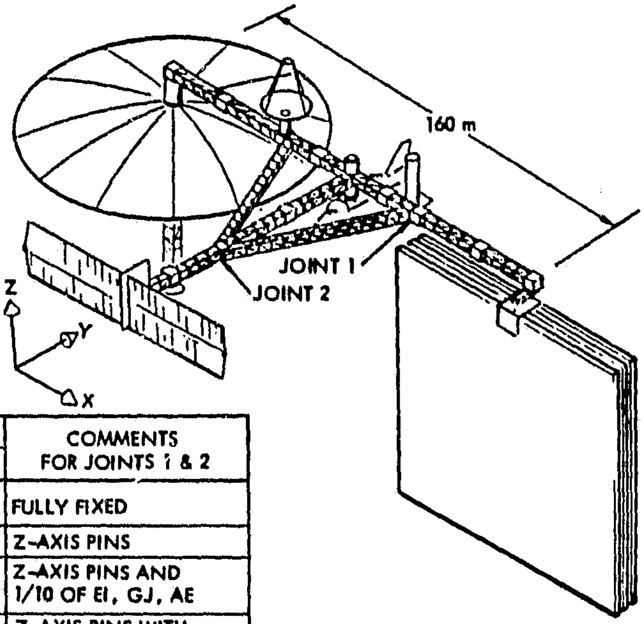
Since the overall variation in platform stiffness is most easily defined by the resulting modal frequencies, modal analyses are performed for the expected diagonal member end joint variations. At all other joints, flexural and torsional moment continuity is maintained.

The table on Figure 1.4-34 illustrates the first four modal frequency variations relative to that of the baseline design. For all of the four cases shown, the stiffness reductions were acceptable.

For other possible configurations:

- Moment and torsion capability of cantilevered members can be provided by fixed joints or appropriate self-locking latches (Figure 6)
- In worst case, EVA attachment of strut or pair of struts is feasible, if necessary.

MODEL:
- JOINT 1 & 2 LENGTHS ARE EACH 2.5 m
- EI & GJ ARE ADOPTED VALUES



CASE	FREQUENCY (Hz)				COMMENTS FOR JOINTS 1 & 2
	FIRST	SECOND	THIRD	FOURTH	
BASE-LINE	0.0431	0.0703	0.0823	0.1181	FULLY FIXED
1	0.0361	0.0658	0.0686	0.1129	Z-AXIS PINS
2	0.0359	0.0651	0.0683	0.1120	Z-AXIS PINS AND 1/10 OF EI, GJ, AE
3*	0.0361	0.0658	0.0686	0.1152	Z-AXIS PINS WITH OFFSET FROM DIAGONAL CENTERLINE
4	0.0351	0.0628	0.0680	0.1085	BALL JOINTS

*OFFSET = 1.5 m

Figure 1.4-34. ASASP Modal Frequency Variation with Diagonal End Joint Design

The third design concern has not been resolved in this study and is applicable to the concepts studied which all utilize clevis joints, folding and/or telescoping joints.

Consider the following illustrative example (Refer to Figure 1.4-35):

- o Consider a cantiliver member 60 m long (38 bays)
- o Suppose ACS thruster shear = 4.5 N
- o Limit longeron load = 2.8 N (one bay from RCS thruster)
- o Limit longeron load = 107 N (38 bays from RCS thruster)
- o For the adopted EI = 2×10^8 Nm^2, and truss depth of 1.26 m the longeron AE = 1.26×10^8 N
- o Elongation of longeron for load of 2.8 N = 0.0000035 cm
- o Elongation of longeron for load of 107 N = 0.00013 cm
- o For a design to twice the depth (same EI), the above values of 0.0000035 and .00013 cm are multiplied by 4, i.e., elongation is 0.000014 and 0.00052 cm

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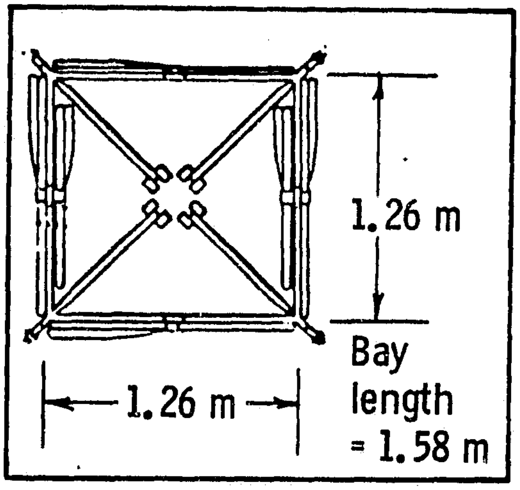

Figure 1.4-35. Model for "Implication of Joint Slop" Analysis

- For both depths, the elongations are very small compared to conventional clevis/pin clearances

- Problem is increasingly more severe with reduced RCS thrust

Hence, for future resolution: (1) To what extent does friction preclude "joint slop?" and (2) What is statistical implication of joint slop in actual design with numerous joints?

The potential implications of "joint slop" to the control system design are:

- It will be of increasing concern in figure control applications where "joint slop" alone results in structural deflections approaching total allowable deflection

- "Joint slop" is highly non-linear phenomenon with strong potential for destabilization and limit cycling

- It will be of increasing concern if "equivalent frequency" (f_e) (Figure 1.4-36) approaches control system bandwidth

The resulting potential control system design implications are:

- Develop accurate "joint slop" model to facilitate controller design

- Typical controller enhancement - nonlinear gain scheduling, compensation and limiters to enhance performance

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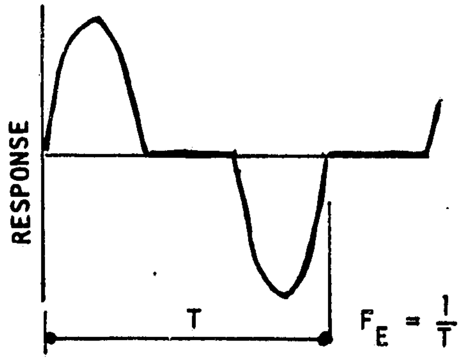

Figure 1.4-36. Equivalent Structure Frequency due to "Joint Slop"

The primary problem is that joint slop introduces a series of distributed non-linearities in the spacecraft dynamics. In general, this will degrade control system performance. It is destabilizing and a potential cause of limit cycling. Special control enhancements can be used to minimize the detrimental effects of joint slop, as indicated above. In general, the problem is tractable if the magnitude of the joint slop can be eliminated or made small relative to the control system pointing accuracy requirement. Also, the control problems are eased with increasing friction levels at the joints.

The preferred solution is to eliminate or negate joint slop by mechanical means (eccentric clevis pin or bolt with an expanding sleeve) or with thermosetting materials at the joints (one bay at a time deployment enhances this possibility). Another alternative is to increase joint friction providing it is not detrimental to deployment.

2. MATERIAL DATA BASE

The material requirements for the manufacture of structural members for large deployable platforms include those needed to produce tension/compression members, tension members only, fittings, hinges, bearings and springs. The materials selected for the data base to satisfy these requirements are those currently available in present technology and those which could be available in the technology for point design by 1986. This philosophy permits the consideration of not only metals and organic composites, but metal matrix composite materials.

This material data base is not intended to permit the detailed optimization needed for the ultimate point designs to be made by 1986. It is intended to supply a sufficiency of general information to allow the material trades required for the generation of preliminary designs. These trades will specify the direction of the ultimate designs for deployable space structures.

All materials selected are suitable for the temperature fluctuations (-200 to 80C) that will be encountered in service in either a low earth or geosynchronous orbit. These temperature excursions can be reduced by the judicious application of thermal control coatings and insulations. Temperature resistance of the materials is, therefore, not a critical consideration. The radiation resistance of the material is important only in the case of geosynchronous orbit. Using a radiation criterion of a dose rate of 6×10^2 rads (Si) per day as the maximum anticipated in geosynchronous orbit, the materials have life expectancies in excess of 5 to 500 years (Reference 13). Since only a few of the materials have low damage thresholds and these are usually restricted to low radiation exposures in the interior of spacecraft, a minimum of thirty-year life is expected for any platform system designed using the materials in the data base.

The data base is contained in Tables 2.2-0 through 2.0-11. Table 2.0-1 provides a glossary of terms. Tables 2.0-2 through 2.0-11 present candidate materials and their pertinent material properties.

Tension and Compression Members

Some typical metals such as aluminum 2219 and beryllium (Table 2.0-2) are included in the data base for comparison with the more advanced materials such as the organic and metal matrix composite materials (Table 2.0-3). Considering the design parameters of low thermal distortion, low cost, and high modulus, the most attractive candidate materials within the state of the art are the epoxy/graphite composites. The metals have a high coefficient of thermal expansion which contributes to excessive thermal distortion of the structure. The polyimide/graphite is state of the art, but it is used only for high temperature applications because it is excessively expensive for general use. The metal matrix composites have all the right properties, but are at present very expensive and the technology is in the experimental stage and may not be available for production by 1986. Therefore, they may have to be relegated to specialty applications where no other material can reasonably suffice. Some new materials that will very probably be available for general application to point designs in 1986 are the very high modulus graphite fibers. These fibers permit the production of structures with a lower

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Table 2.0-1. Glossary of Terms for Data Base

MAIN CHARACTER

A	ACCEPTABLE	M	MEMORY
B	ACCEPTABLE WITH SPECIFIC COUPONS	L	LOSS FACTOR
B_e	BEARING STRENGTH	N/A	NOT APPLICABLE
C	ACCEPTABILITY MUST BE DEMONSTRATED BY TEST OR ANALYSIS	NR	NOT RATED
		R	REFLECTANCE
D	DUCTILITY	σ	SURFACE RESISTIVITY
d_{He}	DIFFUSION OF HELIUM	S	SET POINT
D_{ie}	DIELECTRIC CONSTANT	Sp	SPECIFIC HEAT
D_{is}	DISSIPATION FACTOR	TML	TOTAL MASS LOSS
E	MODULUS	U	UNAVAILABLE
e	ELONGATION	VCM	VOLATILE CONDENSIBLE MATERIAL
F	STRESS	$1/\gamma$	VOLUME RESISTIVITY
f	FATIGUE PROPERTIES	X	UNACCEPTABLE
G	SHEAR MODULUS	α	ABSORBTIVITY
H	HARDNESS	ϵ	EMISSIVITY
K	COEFFICIENT	μ	POISSON's RATIO
k	THERMAL CONDUCTANCE	ρ	DENSITY

SUPERSCRIPT		SUBSCRIPT	
c	COMPRESSION	C	TEMPERATURE
E	MODULUS	e	THERMAL EXPANSION
t	TENSILE	f	FRICTION
u	ULTIMATE	L	LONGITUDINAL
s	SHEAR	T	TRANSVERSE

DIMENSIONS

MPa	- MEGA PASCALS
GPa	- GIGA PASCALS
J/kgK	- JOULES/KILOGRAM-DEGREE KELVIN
W/mK	- WATTS/METER-DEGREE KELVIN
m/mK	- METER/METER-DEGREE KELVIN
kg/m^3	- KILOGRAM/CUBIC METER
m^2/sec-ATM	- SQUARE METERS/SECOND-ATMOSPHERE

coefficient of thermal expansion, a higher thermal conductivity and a high specific rigidity. The two more prominent contenders in the field of high modulus fiber composites are the P75S/934 and P100S/934 graphite/epoxy composites listed in Table 2.0-3.

Fittings

In addition to the conventional materials used in the manufacture of fittings for spacecraft, such as the metals in Table 2.0-4, more unconventional materials (Table 2.0-5) are being considered to obtain both a closer thermal expansion match to the basic structure and a closer match to the modulus. Another major advantage for the use of composite fittings is that it permits the molding of fittings directly into structural members. The type of organic materials most amenable to this type of processing are the thermoplastic composites such as the polysulfone/graphites. There is some development work, however, that needs to be accomplished between now and FY 1986 to reduce the direct molding to structural member process to common everyday practice.

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Table 2.0-2. Metallic Materials for Structural Members

PROPERTIES ① ② \ MATERIALS ②	ALUMINUM 2219	BERYLLIUM
MECHANICAL		
F_L^{tu} (MPa)	427.8	276.0
F_L^{cu} (MPa)	331.0	207.0
F_L^{su} (MPa)	248.4	172.0
E_L^{t} (GPa)	72.5	289.0
E_L^{c} (GPa)	74.5	289.0
G_L (GPa)	27.6	138.0
μ	0.33	0.10
B_e (MPa) e/D = 1.5 e/D = 2.0	655.0 835.0	N/A
PHYSICAL		
Sp (J/kgK)	848.0	1825.0
k (W/mK)	12.1	8.4
K_e ($m/mK \times 10^{-6}$)	22.0	12.3
ε	0.05	0.10
α	0.10	0.55
R	0.90	0.45
ρ (kg/m^3)	2830.0	1850.0
APPLICATIONS	TENSION & COMPRESSION BEARINGS	TENSION & COMPRESSION

① ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS

② ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML

Table 2.0-3. Structural Member Candidate Composite Materials

PROPERTIES \ MATERIAL ①②③	GRAPHITE/EPOXY ①			POLYIMIDE/ GRAPHITE ①	ALUMINUM/ GRAPHITE ①	MAGNESIUM/ GRAPHITE ①
	T300/934	P75S/934	P100S/934			
MECHANICAL						
F_L^{tu} (MPa)	1482.0	999.0	1140.0	863.0	483.0	635.0
F_T^{tu} (MPa)	48.0	32.0	37.0	20.7	48.0	35.0
F_L^{cu} (MPa)	1276.0	328.0	311.0	538.0	642.0	621.0
F_T^{cu} (MPa)	186.0	119.0	U	89.0	104.0	104.0
F_L^{su} (MPa)	69.0	44.0	48.0	41.0	48.0	48.0
F_T^{su} (MPa)	33.0	21.0	23.0	20.0	55.0	U
E_L^t (GPa)	145.0	368.0	428.0	304.0	207.0	324.0
E_T^t (GPa)	10.0	6.0	U	6.2	35.0	21.0
E_L^c (GPa)	133.0	246:0	428.0	254.0	U	U
M (%)	5.0	4.9	U	4.83	24.1	17.3
PHYSICAL						
Sp (J/kgK)	880.0	U	U	879.0	962.9	962.9
k (W/mK)	N/A	N/A	N/A	N/A	233.0	1030.0
k_L (W/mK)	9.0	157.0	520.0	77.83	N/A	N/A
k_T (W/mK)	0.7	1.8	U	1.21	N/A	N/A
K_{eL} (m/mK x 10^{-6})	0.23	-1.04	-1.6	-0.99	1.33	0.55
K_{eT} (m/mK x 10^{-6})	25.2	25.6	26.9	28.8	26.6	28.2
ε	0.85	0.85	0.85	0.85	0.45	0.45
α	0.80	0.80	0.80	0.80	0.45	0.45
R	0.20	0.20	0.20	0.20	0.55	0.55
ρ (kg/m^3)	1600.0	1765.0	1820.0	1580.0	2410.0	1880.0

① ALL COMPOSITE PROPERTIES ARE BASED ON 0° FIBER ORIENTATION
② ALL PROPERTIES MEASURED UNDER AMBINET CONDITIONS
③ ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML

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Table 2.0-4. Metallic Materials for Fittings and Springs

PROPERTIES (1)	MATERIALS (2): CRES 304L-1/8H	MAGNESIUM AZ31B	STEEL 4340	ALUMINUM 2219	BERYLLIUM	TITANIUM (6AL-4V)	302 STAINLESS STEEL	17-7PH STAINLESS STEEL
MECHANICAL								
F_L^{tu} (MPa)	518.0	235.0	1790.0	427.8	276.0	1000.0	587.0	1830.0
F_L^{cu} (MPa)	370.0	110.0	1655.0	331.0	207.0	1034.0	414.0	1103.0
F_L^{su} (MPa)	345.0	159.0	1080.0	248.4	172.0	621.0	366.0	793.0
E_L^t (GPa)	200.0	448.0	200.0	72.5	289.0	116.0	193.0	200.0
E_L^c (GPa)	200.0	448.0	200.0	74.5	289.0	113.0	193.0	206.0
G_L (GPa)	79.4	16.6	75.9	27.6	138.0	42.8	79.4	75.9
μ	0.28	0.35	0.32	0.33	0.10	0.31	0.28	0.28
PHYSICAL								
Sp (J/kgK)	502.0	1004.8	477.0	848.0	1825.0	544.0	502.0	460.5
k (W/mK)	176.0	76.0	38.0	12.1	8.4	7.0	16.25	16.1
K_e (m/mK x 10^{-6})	17.2	25.3	11.3	22.0	12.3	9.4	17.2	10.4
ρ (kg/m^3)	8030.0	1770.0	7840.0	2830.0	1850.0	4430.0	8030.0	7760.0
APPLICATIONS	FITTINGS	FITTINGS	FITTINGS	FITTINGS	FITTINGS	FITTINGS SPRINGS	SPRINGS	SPRINGS

NOTES: (1) ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS.
(2) ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML.

Table 2.0-5. Non-Metallic Materials for Fittings and Springs

(3) PROPERTIES \ MATERIALS (2)	GRAPHITE/EPOXY (1) T300/934	GRAPHITE/EPOXY (1) P75S/934	GRAPHITE/EPOXY (1) P100S/934	POLYIMIDE/GRAPHITE (1)	POLYSULFONE/GRAPHITE (1)
MECHANICAL					
F_L^{tu} (MPa)	1482.0	999.0	11[illegible].0	863.0	1323.0 (4)
F_T^{tu} (MPa)	48.0	32.0 (4)	37.0 (4)	20.7	43.0 (4)
F_L^{cu} (MPa)	1276.0	328.0	311.0	530.0	1139.0 (4)
F_T^{cu} (MPa)	186.0	119.0	U	89.0	166.0 (4)
F_L^{su} (MPa)	69.0	44.0 (4)	48.0	41.0	54.0 (4)
F_T^{su} (MPa)	33.0 (4)	21.0 (4)	23.0 (4)	20.0 (4)	26.0 (4)
E_L^t (GPa)	145.0	368.0	428.0	304.0	119.0 (4)
E_T^t (GPa)	10.0	6.0	U	6.2	4.3 (4)
E_L^c (GPa)	133.0	246.0	428.0	254.0 (4)	U
G_{LT} (GPa)	5.0	4.9	U	4.83	U
μ_{LT}	0.29	0.365	U	0.30	U
PHYSICAL					
S_p (J/kgK)	880.0	U	U	879.0	U
k_L (W/mK)	9.0	157.0	520.0	77.83	9.0 (4)
k_T (W/mK)	0.7	1.8	U	1.21	U
K_{eL} (m/mK x 10^{-6})	0.23	-1.04	-1.6	-0.99	0.23 (4)
K_{eT} (m/mK x 10^{-6})	25.2	25.6	26.9	28.8	U
ϵ	0.85	0.85	0.85	0.85	0.85
α	0.80	0.80	0.80	0.80	0.80
R	0.20	0.20	0.20	0.20	0.20
ρ (kg/m^3)	1600.0	1765.0	1820.0	1580.0	1490.0
APPLICATIONS	FITTINGS SPRINGS			FITTINGS SPRINGS	SPRINGS

NOTES: (1) ALL COMPOSITE PROPERTIES ARE BASED ON 0° FIBER ORIENTATION
(2) ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS
(3) ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML
(4) CALCULATED VALUE FROM BASIC MATERIAL PROPERTIES

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Springs

Table 2.0-4 presents the conventional spring materials normally used in spacecraft. These materials are metallic and have the same shortcomings of excessive thermal expansion as metals proposed for other applications on the deployable platform systems. To obtain a close match of the coefficient of thermal expansion to a graphite composite structure, it is desirable to manufacture the spring out of the same basic materials. Some of the new organic composite materials being considered for the production of springs are presented in Table 2.0-5. These composite springs may not be commercially available by 1986. However, titanium springs can be considered as the closest compatible compromise in the event commercial availability of composites lags the need date.

Flexible Tension Members

Flexible tension members usually consist of metal cables such as the 17-7PH stainless steel presented in Table 2.0-6. This type of construction has two major deficiencies for this application. One, the high coefficient of thermal expansion of the metals makes these cables incompatible with the contemplated graphite/epoxy structure. To further aggravate this condition, the twist in the material can exaggerate the dimensional mismatches that occur with thermal excursions. To reduce, if not eliminate this problem, graphite fiber tapes of monofilaments with flexible organic binders are proposed. A sample of one of these materials is RTV566/graphite and is presented in Table 2.0-6. Most of the properties of the individual components of the composites are known but significant work remains to confirm the potentials of these materials in a combined form as flexible cables.

Table 2.0-6. Materials for Tension Cables

(2) PROPERTIES / MATERIALS (3)	RTV 566/ GRAPHITE (1)	17-7PH STAINLESS STEEL
MECHANICAL		
F_L^{tu} (MPa)	U	1830.0
E_L^t (GPa)	U	200.0
PHYSICAL		
Sp (J/kgK)	U	460.5
k (W/mK)	U	16.1
K_e (m/mK x 10^{-6})	U	10.4
ϵ	0.90	0.30
α	0.85	0.10-0.25
R	0.15	0.90-0.75
ρ (kg/m^3)	U	7760.0
APPLICATIONS	TENSION	TENSION

NOTES: (1) ALL COMPOSITE PROPERTIES ARE BASED ON 0° FIBER ORIENTATION
(2) ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS
(3) ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML

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Electrical Conductors

The primary criterion for electrical conductors, apart from the obvious properties such as specific electrical conductivity, is the ability to coil and uncoil the conductors several times through very tight radii for stowage and deployment in a deployable structure. To this end, copper with its smaller radius of wire for equivalent electrical conductivity, surpasses aluminum. Its superior mechanical properties for like wire gauges surpass silver for this application (Table 2.0-7).

Table 2.0-7. Materials for Electrical Conductors

① PROPERTIES \ MATERIALS ②	ALUMINUM 1100	COPPER	SILVER
MECHANICAL			
F_L^{tu} (MPa)	165.0	207.0	138.0
E_L^t (MPa)	68.2	117.0	73.0
PHYSICAL			
Sp (J/kgK)	958.8	385.2	238.6
k (W/mK)	222.0	7.11×10^5	427.0
K_e (m/mK $\times 10^{-6}$)	23.86	16.74	19.0
ρ (kg/m^3)	2710.0	8920.0	10,500.0
$1/\gamma$ (ohm-cm $\times 10^{-6}$)	2.6548	1.6730	1.50
APPLICATIONS	FLUID LINES ① ELECTRICAL CONDUCTOR	ELECTRICAL CONDUCTOR	ELECTRICAL CONDUCTOR

① ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS

② ALL MATERIALS LISTED MEET S[illegible]-0022 FOR VCM AND TML

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Diffusion Barriers for Fluid Lines

Fluid diffusion barriers (Table 2.0-8) are one application where only a metal can perform effectively for long exposure times. There are a number of candidate materials with little to choose between. The possible exception is the use of the metal matrix composites which technically provide the best of all worlds, low coefficient of thermal expansion, high resistance to gas diffusion, high modulus and low weight. The drawbacks are extreme high cost and the commercial availability by 1986.

Table 2.0-8. Diffusion Barriers for Fluid Lines

MATERIALS (4) / PROPERTIES (3)	17-7PH STAINLESS STEEL	ALUMINUM/ GRAPHITE (3)	ALUMINUM 5052	ALUMINUM 5056	ALUMINUM 1100	TITANIUM (6AL-4V)
MECHANICAL						
F_L^{tu} (MPa)	1830.0	483.0	289.8	414.0	165.0	1100.0
E_L^t (GPa)	200.0	207.0	70.4	71.1	68.2	116.0
PHYSICAL						
Sp (J/kgK)	460.5	962.9	962.9	921.0	958.8	544.0
k (W/mk)	16.1	233.0	138.0	116.6	222.0	7.0
K_e (m/mK x 10^{-6})	10.4	L 1.33 T 26.6	24.1	24.1	23.86	9.4
d_{He} (m^2/sec-ATM)	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE
ρ (k/m^3)	7760.0	2410.0	2690.0	2640.0	2710.0	4430.0
CHEMICAL						
HYDRAZINE	A	A	A	A	A	A
HELIUM	A	A	A	A	A	A
FREONS AS A CLASS	A	A	A	A	A	X (5)
NITROGEN TETROXIDE	A	A	A	A	A	A
APPLICATIONS	FLUID LINES (1)	FLUID LINES (1)	FLUID LINES (1)	FLUID LINES (1)	FLUID LINES (1)	FLUID LINES (1)

NOTES:
(1) THE FLUID LINES MAY BE OVERWRAPPED HYBRIDS OR SINGLE MATERIALS
(2) ALL COMPOSITE PROPERTIES ARE BASED ON 0° FIBER ORIENTATION
(3) ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS
(4) ALL MATERIALS LISTED MEET SP-R-0022 FOR VCM AND TML
(5) CHLORIDES AND CHLORINATED HYDROCARBONS OTHER THAN FREON TF ARE UNACCEPTABLE

Diffusion Barriers-Bellows

Considering the extended exposure times anticipated for flexible portions of the fluid lines the only materials reasonable to consider for this application are metal bellows. The two primary materials currently in use are presented in Table 2.0-9. It is not anticipated that any significant new material will be available for point design by FY 1986.

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Table 2.0-9. Diffusion Barriers for Bellows

PROPERTIES*	MATERIALS	
	321 STAINLESS STEEL	INCONEL 718
MECHANICAL		
F_L^{tu} (MPa)	586.0	1240.0
F_T^{tu} (MPa)	N/A	N/A
F_L^{cu} (MPa)	N/A	N/A
E_L^t (GPa)	193.0	200.0
E_T^t (GPa)	N/A	N/A
μ	0.28	0.31
PHYSICAL		
Sp (J/kgK)	502.0	418.7
d_{He} (sec/m^{2}s)	NEGLIGIBLE	NEGLIGIBLE
ρ (kg/m^3)		
$1/\gamma$ (ohm-cm x 10^{-6})	N/A	N/A
CHEMICAL		
HYDRAZINE	A	A
HELIUM	A	A
FREONS AS A CLASS	A	A
NITROGEN TETROXIDE	A	A
APPLICATIONS	BELLOWS	BELLOWS
*ALL PROPERTIES MEASURED UNDER AMBIENT CONDITIONS		

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Vibration Damping Materials

Currently, there are primarily two significant space-rated materials for vibration damping. These materials are presented in Table 2.0-10. It is not anticipated that any unusual developments will require any unique materials development in this area.

Table 2.0-10. Vibration Damping Materials

PROPERTIES \ MATERIALS	SMERD G.E.	SYNTACTIC FOAM D AIRCRAFT
MECHANICAL		
F_L^{tu} (KPa)	2760.0	2208.0
e (%)	100.0	55.0
Torsional Shear (KPa)	U	1035.0
G (KPa)	U	41,400.0
Tear Strength (N-M)	U	11.3
H (Shore A)	50.0 min.	70.0-85.0
D_{is}	0.162 (1 KHz)	0.12 (1 MHz max.)
PHYSICAL		
VCM (%)	0.02	0.10
TML (%)	0.19	0.22
Glass Transition (°K)	U	227.0
APPLICATIONS	VIBRATION DAMPING	VIBRATION DAMPING

Thermal Control Coatings

Some typical space-rated thermal control coatings currently available are presented in Table 2.0-11. This area requires some extended development not only for deployable space structure, but for all spacecraft. Very few materials serve in this capacity and meet the outgassing contamination requirements of NASA specification SP-R-0022. These materials have some problems with adhesion and general in-space deterioration from causes not fully understood.

Table 2.0-11. Thermal Control Coatings

PROPERTIES \ MATERIALS	BLACK CHEM-GLAZE Z306	WHITE S13G/LO (IITRI)
MECHANICAL		
Adhesion (1)	NO EVIDENCE OF LIFTING	NO EVIDENCE OF LIFTING
PHYSICAL		
α	0.94	0.22
ϵ	0.86	0.85
VCM (%)	0.01 (3)	0.03 (2)
TML (%)	0.56 (3)	0.47 (2)
APPLICATIONS	THERMAL CONTROL COATING	THERMAL CONTROL COATING

NOTES: (1) TESTED PER ASTM D-522
(2) FULLY CURED (30 DAYS OR LONGER) AT ROOM TEMPERATURE
(3) CURED 30 DAYS AT ROOM TEMPERATURE

Lubricants

Several space-rated lubricants exist for most of the typical applications that are anticipated. For solid film lubricant applications, there are the molybdenum disulfides such as Molykote 3402 which conforms to MIL-L-3937 and Adrecolube 13 which conforms to MIL-L-46010. There is also a tungsten disulfide solid film lubricant Microseal. Where greases are needed there is a fluorinated material, Braycoat 3L-38RP; and for an oil, a fluorinated material, Brayco 815Z. All these materials meet the requirements for outgassing in SP-R-0022. The materials listed are typical of the many available and are presented for illustration only.

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3. IDENTIFICATON OF TECHNOLOGY DEVELOPMENT NEEDS

Table 3.0-1 presents the special technology needs summary. It summarizes the estimated cost of resolving each new technology need, as well as the estimated calendar time.

Table 3.0-1. Special Technology Needs Summary

POTENTIAL TECHNOLOGY REQUIREMENT (LISTED BY RATING)	ESTIMATED TOTAL COST ($)	SCHEDULE TO ACCOMPLISH (YR)	PRIORITY
• DAMPING CHARACTERISTICS PREDICTION TECHNIQUE	350-400,000	4.0	HIGHEST
• ORBITAL TRANSFER THRUSTER (1335-2225 N THRUST)*	780-960,000	3.5	
• SPACE CHARGE DISSIPATION TECHNIQUES	124-150,000	4.0	
• JOINTS WITHOUT "SLOP"	190-220,000	1.5	
• MINIMIZE STRUCTURAL LOAD AMPLIFICATION	295-340,000	3.5	
• RADIATION-RESISTANT FIBER OPTICS	120-150,000	1.0	MEDIUM
• LIGHTWEIGHT, LOW CTE, HIGH-STRENGTH CLUSTER FITTING	210-240,000	1.7	
• PASSIVE STRUCTURAL DAMPING TECHNIQUES	100-125,000	2.0	
• MICROMETEOROID STRUCTURE DAMAGE	100-150,000	1.5	
• RADIATION-RESISTANT THERMAL CONTROL COATING	95-115,000	0.8	
• HIGH-CAPACITY HEAT PIPE	600-800,000	3.5	LOWEST
• ACTIVE STRUCTURAL CONTROL TECHNIQUES	300-350,000	3.3	
• ADAPTIVE CONTROL TECHNIQUES	340-385,000	2.7	
• LIGHTWEIGHT, HIGH-STRENGTH MEMBER (MALE) FITTING	155-175,000	1.7	
• INSULATED FLEXIBLE COOLANT LINES	190-240,000	2.5	
• LOW CTE, HIGH MODULUS, FLEXIBLE TENSION MEMBER	100-130,000	1.0	
TOTAL	4,930,000 (HIGHEST)	4.0 (LONGEST)	

*THIS TECHNOLOGY REQUIREMENT IS INVALID IF EARTH STORABLE PROPELLANTS ARE CHOSEN FOR THE ORBIT TRANSFER FUNCTION.

The technology needs are presented in three priority groups. The groups are based upon consideration of net effect on performance, hardware level of the problem, type of logic for resolution, level of test simulation required, development test approach, and required hardware interfaces. There is no significance to the order of the items listed within each priority group.

It is pertinent to note that numerous development tests need to be performed pertaining to electrical cable bending, fatigue data, suitability of folding joints, telescoping joints, etc. These were not considered new technology items since the technical approach is based on known methodologies.

The logic diagram shown in Figure 3.0-1 indicates the approach Rockwell used to identify, validate, and estimate the cost of the special technology needs.

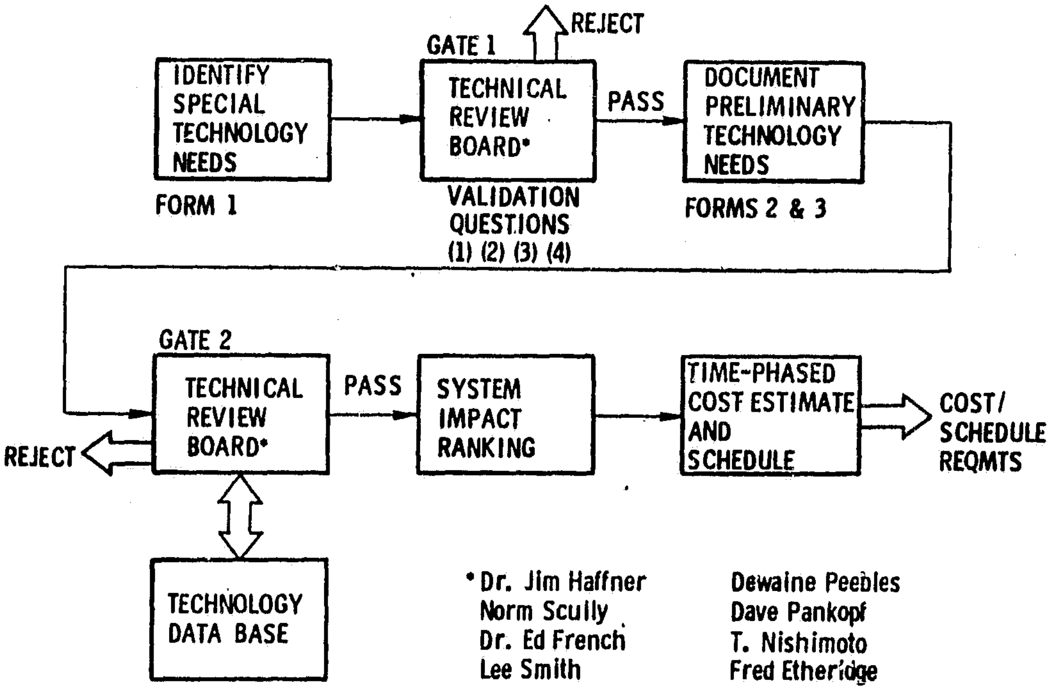

Figure 3.0-1. Logic Diagram—Special Technology Needs

Table 3.0-2 summarizes the 23 potential technology items identified, along with the rationale used to reject seven of the items. As might be expected, the greatest number of items are identified for structures and materials, as well as dynamics and control.

Each of these potential technology requirements was validated by asking the four questions shown in Table 3.0-3. The Technical Review Board (TRB), representing in-depth knowledge in each of the disciplines covered, was able to identify solutions/ongoing R&D activities for seven of the 23 items discussed. The remaining 16 items, which passed Gate 1, were scrutinized further as the TRB collected confirming data. The data were reviewed at the second gate, with confirmation of the 16 primary technology candidates.

Most of the requirements apply to all the structural concepts generated, not just to individual concepts. As an example, radiation-resistant fiber optics apply to all the concepts generated—not just one or two. For this reason, the output of this task is deemed to be inappropriate for inclusion in the selection criteria, but will be a direct input to the Preliminary Test Plan of the Part II study.

A fiscal year schedule for each identified task, including major milestones, was completed for each technology deficiency. Estimated costs were established through discussions with each responsible engineer in each discipline, using a checklist which includes the costs for design and analysis, manufacturing, laboratory testing, major ground tests, flight tests, and (outside) consultation fees.

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Table 3.0-2. Summary of Potential Technology Requirements

POTENTIAL ITEMS	RATIONALE FOR SELECTION
STRUCTURES AND MATERIALS	
• MICROMETEOROID IMPACT STRUCTURE DAMAGE	
• RADIATION-RESISTANT FIBER OPTICS	
• SPACE CHARGE DISSIPATION TECHNIQUES	
• RADIATION-RESISTANT THERMAL COATING	
• LOW CTE, HIGH MODULUS, FLEXIBLE TENSION MEMBER	
• LOW CTE, HIGH MODULUS, COMPOSITE MATERIAL FOR DEPLOYABLE STRUCTURAL ELEMENTS	CURRENTLY UNDER DEVELOPMENT BY G.E. AND CONVAIR
• LIGHTWEIGHT, LOW CTE HIGH-STRENGTH CLUSTER FITTING	
• LIGHTWEIGHT, HIGH-STRENGTH MEMBER (MALE) FITTING	
• COMPOSITE MATERIAL CARPENTER'S HINGE	SHORT-TERM DEVELOPMENT EFFORT
• ZERO-BACKLASH JOINTS	
THERMAL CONTROL	
• INSULATED FLEXIBLE COOLANT LINES	
• LONG-LIFE FLUID PUMP	ORBITER'S RADIATOR FREON PUMP
• HIGH-CAPACITY HEAT PIPE	
• HEAT PIPE INTERCONNECTION TECHNIQUE	CAN RESOLVE BY DESIGN TECHNIQUES
• ROTATING FLUID JOINT FOR ARTICULATING RADIATOR	DESIGN APPROACH DEMONSTRATED DURING FLYING LUNAR EXCURSION EXPERIMENTAL PLATFORM CONTRACT (NAS1-9516, 1970)
UTILITIES	
• TECHNIQUES TO REDUCE BEND RADIUS OF ELEC. CABLES • TECHNIQUES TO REDUCE BEND RADIUS OF NON-POWER CABLES	DESIGN SOLUTIONS ARE WITHIN CURRENT TECHNOLOGY REALM
PROPULSION	
• ORBITAL TRANSFER THRUSTER (1335-2225 N THRUST)	
DYNAMICS AND CONTROL	
• DAMPING CHARACTERISTICS PREDICTION TECHNIQUE	
• ADAPTIVE CONTROL TECHNIQUES	
• ACTIVE STRUCTURAL CONTROL TECHNIQUES	
• PASSIVE STRUCTURAL DAMPING TECHNIQUES	
• MINIMIZE STRUCTURAL LOAD AMPLIFICATION	

Table 3.0-3. Validation Questions

1. IS COMPARABLE WORK BEING CONDUCTED NOW (OR CONTEMPLATED) BY NASA, DOD, OR INDUSTRY?
2. COULD THE REQUIRED NEED DATE BE SATISFIED BY THE ON-GOING TECHNOLOGY RATE/TREND LINE?
3. ARE THERE VIABLE ALTERNATIVES IF THE TECHNOLOGY NEED IS NOT SATISFIED?
4. IS THE SOLUTION TO THE PROBLEM PRIMARILY A SHORT-TERM EFFORT (LESS THAN ONE YEAR)?

The final products of this task are provided at the end of this section. The individual tasks also lend themselves to more accurate milestone schedules and time-phased cost estimates.

The 16 technology items were then prioritized according to the criteria described in Table 3.0-4.

The use of criteria is a method of measuring the impact of a new technology need on deployable platform system performance. The first criterion is a direct measure, while the remaining five are indirect measurements based on the relative difficulty of solving a particular technology deficiency.

The results of the rating are summarized in Table 3.0-5, which indicates that the point count ranges from a high of 4.75, down to 2.00. In those cases where two or more technology deficiencies receive the same numerical total, the same priority rating is assigned. Although this technique is rather arbitrary, the ratings are relative to each other, and a difference of one or two positions is not critical.

Because these rating criteria have only one direct measurement of the impact of a new technology need on system performance, a sensitivity analysis is conducted. The results of this analysis are shown in Table 3.0-6. The upper limit of the first criterion is raised to 3.00 (rather than 1.00) and each priority rating value in this column is multiplied by 3.00 to intensify the first criterion. The results (Table 3.0-6) indicate that roughly half the technology requirements do not change position and the other half moves only one or two positions, i.e., the change is negligible.

Table 3.0-4. Priority Rating Criteria

1. NET EFFECT ON PLATFORM PERFORMANCE AN OVERALL MEASURE OF THE IMPACT THAT A TECHNOLOGY DEFICIENCY HAS ON AN OPERATIONAL DEPLOYABLE PLATFORM DESIGN IS THE NET EFFECT ON PERFORMANCE; AND, BECAUSE THE NET EFFECT INCLUDES ALL TYPES OF TECHNOLOGY DEFICIENCIES, THE TERMS USED TO DESCRIBE THE IMPACT ON THE SYSTEM ARE RATHER GENERAL. OUR NUMERICAL RATING ASSIGNS A VALUE OF 1.00 FOR *SEVERE PERFORMANCE LIMITATION*, WHILE *NO IMPACT* IS ASSIGNED A VALUE OF ZERO.

• SEVERE PERFORMANCE LIMITATION	1.00
• CONSIDERABLE DEGRADATION	0.75
• MILD IMPACT	0.50
• MEASURABLE DEGRADATION	0.25
• NO IMPACT ON PERFORMANCE	0

2. HARDWARE LEVEL OF PROBLEM ONE INDIRECT MEASURE OF THE SYSTEM IMPACT OF A TECHNOLOGY DEFICIENCY IS THE LEVEL (SIZE/COMPLEXITY) OF THE HARDWARE REQUIRED TO RESOLVE THE PROBLEM. GENERALLY SPEAKING, A COMPONENT (SUCH AS A PUMP) REPRESENTS A MUCH LOWER COST IMPACT TO RESOLVE A PROBLEM, COMPARED TO A COMBINED SUBSYSTEMS OR INTEGRATED SYSTEM TEST PROGRAM. THE LATTER NORMALLY REQUIRES MANY ITEMS OF GROUND SUPPORT EQUIPMENT AND KNOWLEDGEABLE TECHNICAL PERSONNEL IN A VARIETY OF DISCIPLINES TO SUCCESSFULLY DIAGNOSE AND CORRECT A SYSTEMS-LEVEL DEVELOPMENT TASK. A COMPONENT, ON THE OTHER HAND, USUALLY REQUIRES A MUCH SMALLER TEAM WITH MORE IN-DEPTH TECHNICAL TRAINING AND A FEW ITEMS OF GROUND SUPPORT EQUIPMENT. OUR NUMERICAL RATING ASSIGNS A VALUE OF 1.00 FOR A SYSTEM-LEVEL PROBLEM BECAUSE OF THE RELATIVELY GREATER COMPLEXITY OF A SYSTEM VERSUS A COMPONENT. A COMPONENT-LEVEL PROBLEM HAS A VALUE OF 0.25.

• COMBINED SUBSYSTEMS/INTEGRATED SYSTEMS	1.00
• SYSTEM/SYSTEM CRITICAL INTERFACE	0.75
• SUBSYSTEM	0.50
• COMPONENT	0.25
• MATERIAL OR METHODS	0

3. TYPE OF LOGIC FOR RESOLUTION THE TYPE OF (COMPUTATIONAL) LOGIC REQUIRED TO RESOLVE A TECHNOLOGY DEFICIENCY IS ALSO AN INDIRECT MEASURE OF SYSTEM IMPACT. CUT-AND-TRY LOGIC IS A FORM OF CURVE FITTING IN A REGION WHERE RELATIONSHIPS ARE NOT WELL DEFINED. THIS METHOD IS TIME-CONSUMING AND REQUIRES EXPERIMENTATION. AN EXAMPLE WOULD BE THE KNOWLEDGE OF FLUID BEHAVIOR IN ZERO GRAVITY IN 1960. AT THE OTHER EXTREME, ANALYSIS OF A PROBLEM USING PROVEN FORMULAS/RELATIONSHIPS AND/OR COMPUTER-STORED PROGRAMS IS RELATIVELY SIMPLE (LEAST COST IMPACT).

• CUT AND TRY	1.00
• DATA EXTRAPOLATION	0.75
• STATISTICAL/EMPIRICAL	0.50
• ANALYSIS/SYNTHESIS	0.25

4. LEVEL OF TEST/SIMULATION REQUIRED TEST LEVELS REQUIRED TO FIND SOLUTIONS TO A PROBLEM ARE ALSO INDICATIVE OF RELATIVE SYSTEM IMPACT. A MAJOR FLIGHT TEST, FOR INSTANCE, IS VERY COSTLY BUT IS SOMETIMES JUSTIFIED AS THE ONLY TEST LEVEL WHICH WILL CREATE ACTUAL COMBINED TEST ENVIRONMENTS WHICH ARE NECESSARY FOR HIGH-CONFIDENCE RESOLUTION. LABORATORY TESTING, ON THE OTHER HAND, TENDS TO BE LESS COSTLY BECAUSE THE TEST TEAM IS GENERALLY SMALL AND THE TEST ENVIRONMENT, EVEN IF CONSIDERED COSTLY, DOES NOT LAST VERY LONG (APPLIED INTERMITTENTLY).

• MAJOR FLIGHT TEST	1.00
• MAJOR GROUND TEST	0.75
• FLIGHT/GROUND PASSENGER	0.50
• LABORATORY	0.25

5. DEVELOPMENT TEST APPROACH THE DEVELOPMENT TEST APPROACH REFERS TO THE PRIMARY METHOD TO BE USED TO OBTAIN THE DESIRED TEST RESULTS. AS AN EXAMPLE, COMBINED TEST ENVIRONMENTS ARE GENERALLY PREFERRED, FROM A TEST POINT OF VIEW, BECAUSE THEY STRESS THE TEST ARTICLE SIMILAR TO ACTUAL USE CONDITIONS AND THEREBY ELIMINATE SOME OF THE UNKNOWNS (THE ADDITIVE EFFECT OF INTERNAL STRESS LEVELS CAUSED BY DIFFERENT ENVIRONMENTS). COMBINED TEST ENVIRONMENTS ARE DIFFICULT TO ACHIEVE AND ARE, THEREFORE, COSTLY TO SIMULATE. A DEMONSTRATION TEST, HOWEVER, IS USUALLY STRAIGHTFORWARD AND, FROM A COST IMPACT STANDPOINT, REPRESENTS RELATIVELY LOW TEST COSTS.

• COMBINED ENVIRONMENTS TESTING	1.00
• DETERMINE SAFETY MARGINS	0.75
• DETERMINE FAILURE MODES	0.50
• DEMONSTRATION	0.25
• GATHER DATA	0

6. OTHER HARDWARE INTERFACES REQUIRED OTHER HARDWARE INTERFACES REQUIRED TO CONTROL, MONITOR, SIMULATE, RECORD, COOL, SUPPORT, POINT, OR TRACK THE TEST ARTICLE ARE ALSO AN INDIRECT MEASURE OF THE IMPACT OF A TECHNOLOGY DEFICIENCY. OTHER HARDWARE INTERFACES REPRESENT CONSIDERABLE COST TO OBTAIN AND OPERATE, OR TO SIMULATE THE EFFECT OF THE INTERFACE. FOR THESE REASONS, *COMBINED SUBSYSTEMS/INTEGRATED SYSTEMS* (AS A HARDWARE INTERFACE) IS THE MOST COSTLY AND, THEREFORE, HAS THE HIGHEST PRIORITY RATING VALUE OF 1.00. *OTHER SUBSYSTEM INTERFACES* REPRESENTS THE LEAST COST IMPACT AND HAS THE LOWEST PRIORITY RATING VALUE OF 0.25.

• COMBINED SUBSYSTEMS/INTEGRATED SYSTEMS	1.00
• OTHER COMPLETE SYSTEMS	0.75
• OTHER PARTIAL SYSTEMS	0.50
• OTHER SUBSYSTEM INTERFACES	0.25

Table 3.0-5. Priority Rating Results

POTENTIAL TECHNOLOGY REQUIREMENTS	PRIORITY RATING CRITERIA: NET EFFECT ON PLATFORM PERFORMANCE	HARDWARE LEVEL OF PROBLEM	TYPE OF LOGIC FOR RESOLUTION	LEVEL OF TEST/ SIMULATION REQUIRED	DEVELOPMENT TEST APPROACH	OTHER HARDWARE INTERFACES REQUIRED	TOTAL	RATING
1. RADIATION-RESISTANT FIBER OPTICS	0.75	0	0.75	0.25	0.75	0.25	2.75	5
2. SPACE CHARGE DISSIPATION TECHNIQUES	1.00	0	0.50	0.50	0.75	0.25	3.50	3
3. RADIATION-RESISTANT THERMAL CONTROL COATING	0.25	0	0.50	0.50	0.75	0.25	2.25	6
4. LOW CTE, HIGH MODULUS, FLEXIBLE TENSION MEMBER*	0.0	0.25	1.00	0.25	0.50	0.50	N/A	8
5. LIGHTWEIGHT, LOW CTE, HIGH-STRENGTH CLUSTER FITTING	0.50	0.25	0.75	0.25	0.50	0.50	2.75	5
6. LIGHTWEIGHT, HIGH-STRENGTH MEMBER (MALE) FITTING	0.25	0.25	0.50	0.25	0.50	0.25	2.00	7
7. ZERO-BACKLASH JOINTS	0.75	0.25	0.50	0.25	0.75	0.50	3.00	4
8. INSULATED FLEXIBLE COOLANT LINES	0.25	0.25	0.50	0.25	0.50	0.25	2.00	7
9. HIGH-CAPACITY HEAT PIPE	0.0	0.25	0.50	0.25	0.50	0.25	N/A	8
10. ORBITAL TRANSFER THRUSTER (1335-2225 N THRUST)	0.75	0.50	0.75	0.50	1.00	0.75	4.25	2
11. DAMPING CHARACTERISTICS PREDICTION TECHNIQUE	0.75	0.75	0.50	1.00	0.75	1.00	4.75	1
12. ADAPTIVE CONTROL TECHNIQUES	0.0	0.50	0.50	0.25	0	0.25	N/A	8
13. ACTIVE STRUCTURAL CONTROL TECHNIQUES	0.0	0.75	0.50	0.25	0.50	0.75	N/A	8
14. PASSIVE STRUCTURAL DAMPING TECHNIQUES	0.50	0.25	0.75	0.25	0.75	0.25	2.75	5
15. MINIMIZE STRUCTURAL LOAD AMPLIFICATION	0.50	0.50	0.50	0.50	0.50	0.50	3.00	4
16. MICROMETEOROID IMPACT	0.75	0.25	0.25	0.75	0.25	0	2.25	6

NOTE: A VALUE OF 0.0 IN THE FIRST COLUMN AUTOMATICALLY PLACES THE REQUIREMENT IN LAST PLACE.
* NOT A SELECTED CONCEPT.

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OF POOR QUALITY

Table 3.0-6. *Sensitivity Trade Results*

POTENTIAL TECHNOLOGY REQUIREMENTS	PRIORITY RATING CRITERIA						TOTAL	RATING
	NET EFFECT ON PLATFORM PERFORMANCE	HARDWARE LEVEL OF PROBLEM	TYPE OF LOGIC FOR RESOLUTION	LEVEL OF TEST/ SIMULATION REQUIRED	DEVELOPMENT TEST APPROACH	OTHER HARDWARE INTERFACES REQUIRED		
1. RADIATION-RESISTANT FIBER OPTICS	2.25	0	0.75	0.25	0.75	0.25	4.25	5
2. SPACE CHARGE DISSIPATION TECHNIQUES	3.00	0	0.50	0.50	0.75	0.25	5.00	3
3. RADIATION-RESISTANT THERMAL CONTROL COATING	0.75	0	0.50	0.50	0.75	0.25	2.75	8
4. LOW CTE, HIGH MODULUS, FLEXIBLE TENSION MEMBER	0.0	0.25	1.00	0.25	0.50	0.50	N/A	11
5. LIGHTWEIGHT, LOW CTE, HIGH-STRENGTH CLUSTER FITTING	1.50	0.25	0.75	0.25	0.50	0.50	3.75	7
6. LIGHTWEIGHT, HIGH-STRENGTH MEMBER (MALE) FITTING	0.75	0.25	0.50	0.25	0.50	0.25	2.50	10
7. ZERO-BACKLASH JOINTS	2.25	0.25	0.50	0.25	0.75	0.50	4.50	4
8. INSULATED FLEXIBLE COOLANT LINES	0.75	0.25	0.50	0.25	0.50	0.25	2.50	9
9. HIGH-CAPACITY HEAT PIPE	0.0	0.25	0.50	0.25	0.50	0.25	N/A	11
10. ORBITAL TRANSFER THRUSTER (1335-2225 N THRUST)	2.25	0.50	0.75	0.50	1.00	0.75	5.75	2
11. DAMPING CHARACTERISTICS PREDICTION TECHNIQUE	2.25	0.75	0.50	1.00	0.75	1.00	6.25	1
12. ADAPTIVE CONTROL TECHNIQUES	0.0	0.50	0.50	0.25	0	0.25	N/A	11
13. ACTIVE STRUCTURAL CONTROL TECHNIQUES	0.0	0.75	0.50	0.25	0.50	0.75	N/A	11
14. PASSIVE STRUCTURAL DAMPING TECHNIQUES	1.50	0.25	0.75	0.25	0.75	0.25	3.75	7
15. MINIMIZE STRUCTURAL LOAD AMPLIFICATION	1.50	0.50	0.50	0.50	0.50	0.50	4.00	6
16. MICROMETEOROID IMPACT	2.25	0.25	0.25	0.75	0.25	0	3.75	7

NOTE

A VALUE OF 0.0 IN THE FIRST COLUMN AUTOMATICALLY PLACES THE REQUIREMENT IN LAST PLACE.

NEW TECHNOLOGY NEED

TASK TITLE: Damping Characteristics Prediction Technique

JUSTIFICATION:

A highly damped structural system will minimize any requirement to augment motion with the attitude control function. A low damped structural system may require augmented attitude control (to meet pointing accuracy) as well as a variable gain function. The latter situation would be more complicated (less reliable) and weigh more than the first situation. The ability to accurately predict the degree of damping may alleviate a complicated attitude control function and avoid an overkill with passive damping techniques.

TECHNICAL OBJECTIVES:

Develop a method of analysis (verified by test data) which will predict the damping characteristics of deployable structures containing extensive utility lines and cables in zero gravity.

TASK DESCRIPTIONS:

1. Develop a modeling technique for utility lines and cables which are secured to basic structure in a variety of ways (various damping coefficients).

2. Verify the above modeling characteristics by free-free modal testing of representative structures and utilities.

3. Verify the above modeling characteristics by space flight testing representative structures and utilities (zero-gravity mode).

4. Update the analysis technique as required.

NEW TECHNOLOGY NEED

TASK TITLE: ORBITAL TRANSFER THRUSTER (1335 TO 2225 N THRUST)*

Justification

Many light-weight structure concepts will be constrained to very low acceleration forces (thrust to weight ratio) during transfer to geosynchronous orbit. A new thruster, in the 1335 to 2225 N thrust range, utilizing liquid oxygen and liquid hydrogen, needs to be developed for this application.

Technical Objective

Design and demonstrate reliable operation of an orbital transfer thruster in the 1335 to 2225 N thrust range, pump fed.

Task Description

Conduct detail design, analysis, manufacture, and development tests to verify performance and reliable operation of an orbital transfer thruster in the 1335 to 2225 N thrust range. A soft thrust buildup transient is a key requirement (5 to 15 second rise time), which may require a variable area injector, a sequenced injector design, or variable-speed, motor-driven pumps.

*This technology requirement is invalid if earth storable propellants are chosen for the orbital transfer function.

NEW TECHNOLOGY NEED

TASK TITLE: SPACE CHARGE DISSIPATION TECHNIQUES

JUSTIFICATION

The space environment, particularly at geosynchronous altitudes, is known to cause a spacecraft to accumulate a differential charge as high as 20 kilovolts when a magnetic disturbance (called a substorm) occurs in its vicinity. Large arcs, or the corona discharge so produced, radiate large-amplitude fast-rise-time electromagnetic pulses that can be detrimental to circuits and circuit piece-parts.

While there are active mechanisms to reduce surface charging (electron gun, heated filament, or a plasma source), long-term, high-reliability requirements suggest a high secondary electron emission material. This area has not been fully explored to date and holds the promise of a passive, long-life, highly reliable method to minimize surface charging of the system.

TECHNICAL OBJECTIVES

- Investigate and recommend, based on simulated environmental testing, a surface coating which will produce secondary electron emissions (or luminescence) at relatively low-voltage spacecraft charging levels.
- Develop a suitable application process for the above recommended coating, based on laboratory testing and verification methods.
- Verify, by space testing, that the basic theory of reducing charge voltage levels, through secondary yield or photon emission of the coating, actually works.

TASK DESCRIPTION

Part I will investigate, by means of laboratory testing, various coating materials which electroluminesce or produce high secondary yields at relatively low-voltage platform charging levels.

The purpose of this task is to demonstrate the above effect by subjecting a test structure, coated with a suitable material, to a simulated electrostatic charge.

Part II will concentrate on developing a suitable method of applying the recommended coating. Laboratory application tests will be conducted, and methods will be developed which verify uniformity and adhesion.

Part III will verify the basic theory of the new coating by space testing. A spacecraft (an ATS geosynchronous vehicle) will be coated with the new material and be suitably instrumented (a harness noise monitor and high-voltage-charge-accumulation monitor) to detect possible static charge buildup and arcing.

NEW TECHNOLOGY NEED

TASK TITLE: Joints without Slop

JUSTIFICATION:

Stiffness of the deployable platform structure will be lost if backlash is present in the joints. Also, the predictability of structural performance characteristics will be imperiled.

TECHNICAL OBJECTIVES:

Develop a zero-backlash joint, suitable for deployable platform designs and operational environments.

TASK DESCRIPTIONS:

1. Perform design and analysis to investigate and select several promising approaches, both passive and active concepts.

2. Manufacture several joints of each design selected above.

3. Perform laboratory test program to verify zero backlash for all operational (load) conditions and environments. Select the design approach which best meets the performance requirements.

NEW TECHNOLOGY NEED

TASK TITLE: Minimize Structural Load Amplification

JUSTIFICATION

Several approaches are possible which will reduce load amplification factors for lightweight structures, namely: (1) soft transient thrust buildup of RCS thrusters (0.1 to 0.25 lb thrust, 10-20 second rise time); and (2) optimum pulsing of RCS thrusters to deadbeat the structural response of the system. The most cost-effective approach must be defined and demonstrated because a structural load amplification of two will require structural members with greater stiffness (more weight) for the deployable platform system.

TECHNICAL OBJECTIVES

Select the most cost-effective approach to minimize structural load amplification for lightweight deployable space platform structures.

TASK DESCRIPTIONS

Two approaches are presented. The optimum pulsing approach is shown to be the most cost-effective method and will be adopted as the baseline method to minimize load amplification.

1. Develop and demonstrate an RCS thruster (0.1 to 0.25 lb thrust level) which has a 10-20 second rise time.

2. Define a suitable method (hardware and mechanization) for optimum pulsing of current design RCS thrusters, such that the structural response of the system does not lead to appreciable load amplification factors. Include verification testing using representative structure and utilities in a space flight mode.

NEW TECHNOLOGY NEED

TASK TITLE: RADIATION RESISTANT FIBER OPTICS

JUSTIFICATION:

Quartz fiber optics are subject to color center formations, called opaquing (degrading effect to light transmission), caused by radiation impingement in a vacuum. The Quartz fiber space lattice is composed of silicon and oxygen atoms which have a one-to-one correspondence (single bond) between each element. Double bonds are formed within the lattice structure (color is added) when radiation (electrons, protons, or photons) displaces an oxygen molecule, thereby destroying one of the single bonds.

Shielding would add considerable mass. The high density material required would be a source of secondary emissions (Bremsstrahlung radiation, produced by the impact of electrons, protons, or photons) which can be more damaging (because these are harder to shield against) than the original radiation.

Two approaches to resolve this problem are presented. The approach which shows the most promise, at the end of the development period, will be adopted as the baseline method for fiber optic applications.

TECHNICAL OBJECTIVES:

1. Determine the possibility that organic materials (such as Acrylic), with higher energy linkage between bonds, are more resistant to galactic radiation and will maintain high transmissibility of light.

2. Determine a suitable method of annealing the Quartz fibers with heat, applied intermittently.

TASK DESCRIPTIONS:

1. Select the most promising organic materials which possess suitable light transmission characteristics.

   Conduct a laboratory testing program to determine the amount of resistance to simulated galactic radiation of each material selected above.

   Based on the above test results recommend an organic material suitable for 10-year life in geosynchronous orbit.

2. Investigate possible methods of annealing Quartz fibers with heat. Conduct laboratory tests to verify that annealing restores the single bonds between elements (no degradation).

NEW TECHNOLOGY NEED

TASK TITLE: Lightweight, low CTE, High-Strength Cluster Fitting

JUSTIFICATION:

Without a low CTE cluster fitting, the effective CTE for the overall deployable platform design could be appreciably higher and result in increased thermal deformation.

TECHNICAL OBJECTIVES:

Develop a technique for producing multi-directional cluster fittings which have the following characteristics: (1) low CTE in each of the projected (longitudinal) directions; (2) close-tolerance clevis pin holes; (3) high bearing allowables at the clevis pin holes; (4) low overall weight; and (5) high strength across the fitting.

TASK DESCRIPTIONS:

1. Manufacture several cluster fittings using different ratios of resin to fiber, different fibers, mixes of fibers (glass/carbon), as well as combination techniques (partial hand layup, partial injection mold). Consider metal inserts for clevis pin holes.

2. Perform a laboratory test program to evaluate longitudinal CTE, bearing stress limits, weight, load capacity across the fitting. Select the manufacturing methods and material which best meet the characteristics desired.

3. Develop a process specification to be used to produce multi-directional cluster fittings.

NEW TECHNOLOGY NEED—CONTINUATION SHEET

TASK TITLE: PASSIVE STRUCTURAL DAMPING TECHNIQUES

JUSTIFICATION. The visco-elastic material currently used to attenuate structural response is very heavy and, therefore, would not be suitable for application to the entire deployable platform. A judicious application to each pinned joint, to cause the joint to respond more like a cantilevered beam, needs to be investigated and design solutions verified by testing.

TECHNICAL OBJECTIVES. Develop passive structural damping techniques suitable for the pinned joints of large deployable space platform structures.

TASK DESCRIPTIONS

1. Design studies to determine best means of applying visco-elastic material to pinned joints.
2. Laboratory test program to verify attenuation of structural response (pinned joints respond more like cantilevered beam).
3. Establish a design standard for applying visco-elastic material to pinned joints for deployable space platforms.

NEW TECHNOLOGY NEED

TASK TITLE: Micrometeoroid Impact Structure Damage

JUSTIFICATION

The probability of micrometeoroid impact upon the structural members in a deployable truss is sufficiently significant to be of concern to the structural integrity. Presently, there is virtually no data descriptive of the impact damage upon structural tubes of circular, square, or rectangular cross sections fabricated from composite materials.

TECHNICAL OBJECTIVE

To develop analytical method (confirmed by testing) for prediction of micrometeoroid impact damage to circular, square, and rectangular tubes of graphite epoxy and aluminum construction.

(1) Develop analytical methodology, using existing techniques to maximum extent possible, to predict structural damage of candidate constructions.

(2) Manufacture nine representative sections of graphite epoxy tubing and nine representative sections of aluminum tubing. Two tubing shapes will be made (nine round, and nine square).

(3) Characterize each different tubing shape (round, square) for compressive strength.

(4) Test six sections of each different tubing shape for hypervelocity impact, using glass beads with a mass of one gram (or less) to produce velocities of 7 km/sec or more. Three sections are to be hit on centerline, and three sections hit one inch off centerline.

(5) Retest each tubing section (which was penetrated) for strength characteristics as described in item (2) above.

(6) Correlate analytical predictions with test data and mobility prediction techniques, as required.

NEW TECHNOLOGY NEED

TASK TITLE: Radiation-Resistant Thermal Control Coating

JUSTIFICATION

The white pigments used in thermal control coatings (such as zinc oxide and titanium dioxide) tend to increase in solar absorptivity in the space environment. The cause is believed to be a combination of ultra violet and entrapped radiation, causing the loss of a small percentage of oxygen or water in the pigment. Most white paints will slowly discolor because of color centers formed (caused by the loss of oxygen or water). This phenomenon causes spacecraft temperature to increase and could imperil the operation of temperature-sensitive components using structure as a heat sink.

TECHNICAL OBJECTIVE

Verify that a new pigment (such as zinc orthotitanate), with an appropriate binder, will not degrade significantly over the ten-year life of the space platform in geosynchronous orbit.

TASK DESCRIPTION

Conduct laboratory tests on new pigments (such as zinc orthotitanate), with an appropriate inorganic binder, to determine the amount of degradation to be expected in a space environment over a ten-year period. Recommend a new pigment and binder combination for long-term space applications.

NEW TECHNOLOGY NEED

TASK TITLE: HIGH CAPACITY HEAT PIPE

Justification

High capacity heat pipes, in the range of 20 KW-meters and above, have not been developed and demonstrated.

Technical Objective

Develop and demonstrate a high capacity heat pipe.

Task Description

Develop and demonstrate a high capacity heat pipe design (20 KW-meters and above), suitable for space platform application.

NEW TECHNOLOGY NEED

TASK TITLE: Active Structural Control Techniques

JUSTIFICATION:

This technology will permit system performance requirements to be achieved in the presence of relaxed structural stiffness, frequency, damping, and alignment requirements.

TECHNICAL OBJECTIVES:

Develop an active structural control technique for vibration suppression and shape control (including use of distributed sensors and actuators) in order to reduce requirements for structural stiffness, frequencies, and damping.

TASK DESCRIPTIONS:

1. Develop methodology.
2. Synthesize active structural control approach for space platform.
3. Simulate and evaluate sensitivity to "real world" errors and develop hardware requirements.
4. Develop sensor and actuator requirements.
5. Verify sensor and actuator performance by space flight-testing representative structure and utilities.

NEW TECHNOLOGY NEED

TASK TITLE: Adaptive Control Techniques

JUSTIFICATION

Large tolerances on structural dynamic parameters, from preflight analytical estimates, will not permit achievement of a high-performance controller. Ground testing of very large structures is difficult and expensive. Classical frequency separation criteria impose unnecessarily severe structural dynamic requirements.

TECHNICAL OBJECTIVES

Development of an in-flight dynamic mode identification technique in order to reduce tolerances of structural dynamic parameters. These data will permit adaptive readjustment of the controller to achieve higher stability/performance and/or reduction in structural bending stiffness and frequency requirements.

TASK DESCRIPTIONS

(1) Develop methodology.

(2) Synthesize adaptive approach for space platform application.

(3) Simulate system and evaluate sensitivity to "real world" errors.

(4) Formulate new criteria for structural dynamic design requirements.

(5) Verify in-flight dynamic mode identification technique by ground testing representative structure and utilities.

NEW TECHNOLOGY NEED

TASK TITLE: Lightweight, High-Strength Member (Male) Fitting

JUSTIFICATION:

Separate (male) fittings will weigh more (and require more volume) than a fitting which is integral with the structural member. Packaging efficiency of the deployable platform may be degraded.

TECHNICAL OBJECTIVES:

Develop a technique for producing lightweight, high-strength member (male) fittings as an integral part (extension) of the structural compression member.

TASK DESCRIPTIONS:

1. Manufacture several member (male) fittings by a layup technique, transitioning from a circular (tube) member to a flat tongue. Consider the addition of different fibers in the male fitting, as well as metal inserts for the clevis pin hole.

2. Laboratory test program to evaluate compression/tension limits and bearing stress limits. Select the manufacturing methods and materials which best meet the desired characteristics.

3. Develop a process specification to be used to produce integral (male) member fittings.

NEW TECHNOLOGY NEED

TASK TITLE: INSULATED FLEXIBLE LINES

Justification

Insulated fluid lines, currently on the market, are not flexible enough to bend to a small radius. Those designs available which will bend to a small radius are quite bulky and require too much room when stowed in a deployable structure.

Technical Objective

Develop a new concept for insulated flexible lines.

Task Description

Design and develop a new concept for an insulated flexible line which is not bulky and will permit bending to a small radius.

NEW TECHNOLOGY NEED

TASK TITLE: Low CTE, High Modulus, Flexible Tension Member

JUSTIFICATION:

The only high-strength tension member which is quite flexible and currently available is stranded wire cable. Wire cable, however, has a substantially higher coefficient of thermal expansion compared to the anticipated composite structure. Net result would be a loss of tension in sunlight and excessive tension in the earth's shadow. The structure's shape, pointing accuracy, and load-carrying capacity would be severely degraded.

TECHNICAL OBJECTIVES:

Develop a high-strength tension member which will bend to a small radius, have an extremely small coefficient of thermal expansion, and not degrade in the space environment.

TASK DESCRIPTIONS:

Laboratory test program to determine the best composite material (such as P100S graphite fibers, imbedded in an elastomeric matrix material) which will bend to a small radius, have an extremely small coefficient of thermal expansion, and will not degrade in a space environment.

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4. CONCEPT SELECTION

This section describes the process used to select the most suitable concept from the eight candidate building block designs, described in Section 1. This process identified Concepts 6 (Figure 1.4-17) and 8 (Figure 1.4-20) as together being the designs that best satisfy the major criteria tabulated in Table 4.1-1. The major reasons for identification of Concepts 6 and 8 are enumerated as follows:

- The open-square shape of Concept 8 has the best growth potential for utilities and can accommodate (in trays) up to twice the adopted study requirements (increased utilities requirements are typical with program maturity).

- For significantly reduced utilities requirements (mountable on longerons) Concept 6 is adequate and is a simpler structural design than Concept 8.

- The two designs together can satisfy the LEO/GEO platform requirements with common concepts of housing, adapter, deployment mechanization and building-block to building-block attachment designs.

- Redundancy for meteoroid impact survival (if necessary) is available.

- The square housing is most suitable for mounting of payloads, subsystems, propulsion modules, and mounting ports.

- The square housing is most amenable to support in the orbiter cargo bay.

The identification of Concepts 6 and 8 was accomplished through review of summary tables (4.10-1 through 4.10-5). These tables encompass the major criteria of Table 4.1-1.

At the conclusion of this selection process, the best features of Concepts 6 and 8 were configured into a new concept called 6A, (Figure 4.11-2). Concept 6A is the same design as Concept 6 except the longerons are folded at 30° rather than 45° (Drawing 42712-29, Volume II), clearing the center of the square frame for installation of utility trays such as are shown for Concept 8 (Drawing 42712-025, Volume II). The overall features of this design are summarized as follows:

- Building-block approach for automatic deployment of platform systems.

- The square-shaped truss is most suitable for inter-building-block attachments; mounting of payloads, docking ports, propulsion modules, etc.

- Circular tubes for all truss members provide minimum cost construction with use of graphite composite construction.

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- Trays for mounting of adopted complement of utilities provide ease of initial installation, repair, and replacement during total ground fabrication period with minimum truss structural design constraints imposed by utilities integration
- Small complements of utilities can be mounted directly onto the longerons (design reduces to Concept 6).
- Square-shaped housing with reciprocating deployment mechanism
- Bay-by-bay deployment to facilitate identification of deployment problem (in the event this occurs).
- Rail system for root strength during deployment permits orbiter berthing and orbiter VRCS firing, if necessary.
- Adapters for mounting of payloads with automatic electrical connector interface.
- Payloads and propulsion modules attached using RMS.

Further detail concerning the selection process is provided in the remainder of this section.

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4.1 MAJOR CRITERIA

The criteria used in the selection are listed in Table 4.1-1. Many other criteria were included and then rejected, since there was no difference among the concepts insofar as these criteria are concerned. For example, one early criterion was the ability to deploy in a straight line. All of the concepts have this characteristic, hence that particular criterion was eliminated.

An explanation of each criterion together with the approach to grading the concepts is provided in Sections 4.3 through 4.9.

Table 4.1-1. Major Criteria Used in the Selection Process

1. DESIGN VERSATILITY (WITH DISTINCTIONS BETWEEN LEO AND GEO) OF STRUCTURAL CONCEPT • Accommodation of adopted power and data utilities requirements • Accommodation of reduced power and data utilities requirements • Accommodation of fluid utilities: Two 2-cm lines (or equivalent) • Satisfaction of adopted strength and stiffness requirements • Satisfaction of strength and stiffness requirements that are each 1/10 of the adopted values • Satisfaction of the adopted strength requirement and 10 times the adopted stiffness requirement • Platform construction • Accommodation of aluminum and graphite composite materials
2. COST OF TOTAL BUILDING BLOCK IN GENERIC PLATFORM • Launch cost • Fabrication cost • Orbit transfer to GEO • Technology development differential (negligible)
3. THERMAL STABILITY OF STRUCTURAL CONCEPT
4. METEOROID IMPACT SUITABILITY OF STRUCTURAL CONCEPT
5. RELIABILITY OF DEPLOYMENT (BUILDING BLOCK) • Basic truss structure • Docking port structure • Housing • Materials variation • Adapter • Mechanization
6. PREDICTABILITY OF PERFORMANCE OF STRUCTURAL CONCEPT
7. INTEGRATION SUITABILITY OF BUILDING BLOCK

4.2 METHODOLOGY

The eight concepts are graded by an allocation of points. Points are allocated to the concepts in two ways:

- o Qualitative data are converted to points judgmentally.
- o Quantitative data are converted to points using a linear system as shown in Figure 4.2-1. Regarding the line marked "baseline evaluation" the most desirable concept is awarded 100% points and the least desirable concept is awarded 50% points. The other concepts are graded on a linear basis between the two extremes. This method was used for all the tables shown up to and including Tables 4.10-1 and -2.

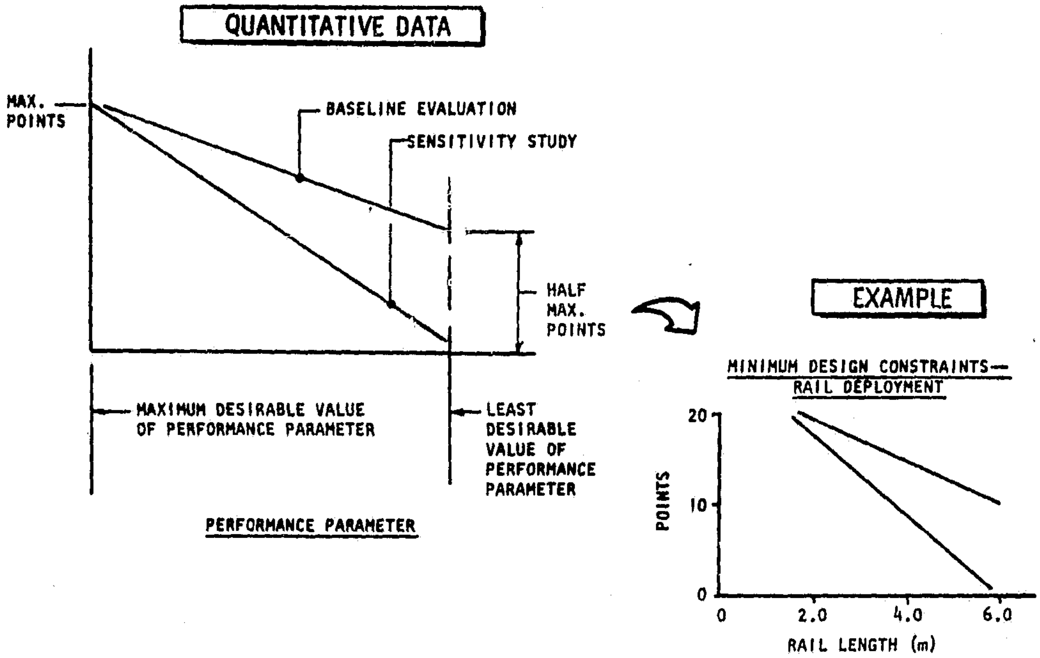

Figure 4.2-1. Point Evaluation Methodology

Another approach is shown on Figure 4.2-1, but uses the line marked "sensitivity study". The only difference is that zero % points are awarded to the least desirable concept instead of 50%. This method was used in compiling summary tables (4.10-3 and -4). Both the "baseline study" and the "sensitivity study" are incorporated into this selection process.

There is only one important distinction between LEO and GEO in terms of concept selection, and that is mass. Other differences between LEO and GEO are listed below, but do not affect concept selection:

- o Transfer to GEO introduced a somewhat higher loading regime, but not sufficient for distinction in trade
- o No significant differences observed in flexural and torsional stiffness requirements

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- o No differences (significant to concept selection) observed in
  - Utilities requirements (function of payloads)
  - Meteoroid environment
  - Thermal environment
  - Servicing and maintenance

4.3 DESIGN VERSATILITY OF STRUCTURAL CONCEPT

The grading of the eight buiding-block concepts for this criterion is shown on seven tables:

- o Table 4.3-1 - Electrical Accomodations (GEO)
- o Table 4.3-2 - Electrical Accomodations (LEO)
- o Table 4.3-3 - Fluid Utilities Accommodations (LEO and GEO)
- o Table 4.3-4 - Structural Materials Variation (LEO and GEO)
- o Table 4.3-5 - Strength & Stiffness Accommodations (GEO)
- o Table 4.3-6 - Strength & Stiffness Accommodations (LEO)
- o Table 4.3-7 - Platform Construction (LEO and GEO)

The following notes are intended to explain the rationale behind the grading and points allocation of Tables 4.3-1 through 4.3-7.

For Tables 4.3-1 through 4.3-3:

- o Mass implications apply to the mass of trays, clips, and shoes, etc., necessary to support utilities.
- o The high electrical requirement is the baseline adopted requirement.
- o The low electrical requirement consists of eight No. 4 lines and 1/8 the complement of adopted data lines.
- o The values assigned are based on accommodation of utilities by the utilities installation designs tabulated as follows:

CONCEPT	LOW REQUIREMENT	HIGH REQUIREMENT	FLUIDS
1	TRAYS	TRAYS	TRAYS
2	TRAYS	TRAYS	LOOPS
3	COILED	COILED	LOOPS
4	LONGERONS	LOOPS	LOOPS
5	TRAYS	TRAYS	TRAYS
6	LONGERONS	LOOPS	LOOPS
7	TRAYS	TRAYS	TRAYS
8	TRAYS	TRAYS	TRAYS

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Tables 4.3-1 and 4.3-2 address the comparative capability of the designs to accommodate the electrical utilities, respectively, for GEO and LEO applications. The following clarification of the evaluation parameters is presented:

- Reliability for Space Deployment

  This parameter addresses the likelihood of successfully moving the electrical lines from the stowed to the deployed position. Retraction is not a factor. Lines which are mounted on trays or on the longerons are thought to be the most reliable. Looped lines are similarly reliable for small quantities of lines, but slightly less reliable for large bundles. Coiled lines are judged to be the least reliable.

- Ease of Ground Deployment and Checkout

  This section addresses the ease/difficulty of extending and retracting the electrical utilities several times in a 1-g environment. Factors included in the assessment are accessibility and the necessity for manual resetting of the lines for retraction.

Table 4.3-1. Electrical Accommodations (GEO)

ELECTRICAL UTILITIES ACCOMMODATION																			
CONCEPT	LOW REQUIREMENT									HIGH REQUIREMENT									TOTAL LOW AND HIGH REQUIREMENT
	RELIABILITY SPACE DEPLOYMENT		EASE OF GROUND DEPLOYMENT & CHECKOUT		SUITABILITY TO LAUNCH ENVIRONMENT		MASS* IMPLICATION		TOTAL LOW REQUIREMENT	RELIABILITY SPACE DEPLOYMENT		EASE OF GROUND DEPLOYMENT & CHECKOUT		SUITABILITY TO LAUNCH ENVIRONMENT		MASS* IMPLICATION		TOTAL HIGH REQUIREMENT	
	RANK	PTS	RANK	PTS	RANK	PTS	Δ MASS	PTS		RANK	PTS	RANK	PTS	RANK	PTS.	Δ MASS	PTS		
	MAX POINTS 20		MAX POINTS 10		MAX POINTS 20		MAX POINTS 20		MAX POINTS 70	MAX POINTS 20		MAX POINTS 10		MAX POINTS 20		MAX POINTS 20		MAX POINTS 70	MAX POINTS 140
1	1	20	3	9	3	18	.14	10	57	1	20	1	10	1	20	.68	10	60	117
2	1	20	3	9	3	18	0	20	67	1	20	1	10	1	20	0.1	19	69	136
3	8	15	8	4	8	10	0	20	49	8	15	8	4	6	12	0	20	51	100
4	1	20	1	10	1	20	0	20	70	6	18	6	6	7	6	.18	18	48	118
5	1	20	6	7	3	18	.14	10	55	1	20	4	8	1	20	.64	11	59	114
6	1	20	1	10	1	20	0	20	70	6	18	6	6	7	6	0.1	19	49	119
7	1	20	6	7	3	18	.14	10	55	1	20	4	8	1	20	.64	11	59	114
8	1	20	5	8	3	18	0.0	20	66	5	19	3	9	1	20	0.3	15	63	129

NOTES:

* KG X 10^{-3}

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Table 4.3-2. Electrical Accommodations (LEO)

CONCEPT	ELECTRICAL UTILITIES ACCOMMODATION													
	LOW REQUIREMENT							HIGH REQUIREMENT							TOTAL LOW AND HIGH REQMT
	RELIABILITY SPACE DEPLOYMENT		EASE OF GROUND DEPLOYMENT & CHECKOUT		SUITABILITY TO LAUNCH ENVIRONMENT		TOTAL LOW REQMT	RELIABILITY SPACE DEPLOYMENT		EASE OF GROUND DEPLOYMENT & CHECKOUT		SUITABILITY TO LAUNCH ENVIRONMENT		TOTAL HIGH REQMT	
	RANK	PTS	RANK	PTS	RANK	PTS		RANK	PTS	RANK	PTS	RANK	PTS		
	MAX POINTS 20		MAX POINTS 10		MAX POINTS 20		MAX PTS 50	MAX POINTS 20		MAX POINTS 10		MAX POINTS 20		MAX PTS 50	MAX PTS 100
1	1	20	3	9	3	18	47	1	20	1	10	1	20	50	97
2	1	20	3	9	3	18	47	1	20	1	10	1	20	50	97
3	8	15	8	4	8	10	29	8	15	8	4	6	12	31	60
4	1	20	1	10	1	20	50	6	18	6	6	7	6	30	80
5	1	20	6	7	3	18	45	1	20	4	8	1	20	48	93
6	1	20	1	10	1	20	50	6	18	6	6	7	6	30	80
7	1	20	6	7	3	18	45	1	20	4	8	1	20	48	93
8	1	20	4	8	3	18	46	5	19	3	9	1	20	48	94

o Suitability to Launch Environment

This parameter subjectively accounts for the degree of support provided to the utilities during launch by the utilities installation systems (longerons, trays, clips, shoes). Electrical utilities mounted on the longerons are best, with trays second, and the remaining concepts last. The points allocated are a judgmental estimate of the relative difference between the designs.

o Mass Implication

This parameter quantitatively accounts for the GEO transfer cost implication of the utilities support system mass. The masses shown represent the delta mass above that of the minimum value.

Table 4.3-3 compares the comparative capability of the designs to accommodate the fluid utilities. The following clarification of the evaluation parameters is presented.

- o Minimum Bend Radius

  A large bend radius in the fluid lines is judged to be better than a small bend radius. A small bend radius has higher stress and takes more force to fold.

- o The notes for Tables 4.3-1 and 4.3-2 apply to the other parameters shown.

Table 4.3-3. Fluid Utilities Accommodation (LEO and GEO)

CONCEPT	FLUID UTILITIES ACCOMMODATION								
	RELIABILITY SPACE DEPLOYMENT		EASE OF GROUND DEPLOYMENT & CHECKOUT		SUITABILITY TO LAUNCH ENVIRONMENT		MINIMUM BEND RADIUS		
	RANK	PTS	RANK	PTS	RANK	PTS	RADIUS (CM)	PTS	TOTAL
	MAX POINTS 20		MAX POINTS 10		MAX POINTS 20		MAX POINTS 20		MAX PTS 70
1	1	20	1	10	1	20	12	15	65
2	5	19	5	8	5	18	20	20	65
3	8	10	6	7	6	16	20	20	53
4	5	19	8	7	7	6	14	14	46
5	1	20	3	9	1	20	4	10	59
6	5	19	6	7	7	6	14	18	50
7	1	20	3	9	1	20	4	10	59
8	1	20	1	10	1	20	3.1	10	60

Table 4.3-4 compares the versatility of the candidate structural designs to use either aluminum or composite materials (graphite epoxy or metal matrix). The judgemental evaluation is based upon the degree of static determinacy of the structure. The maximum points are assigned to statically determinate structures. A pure statically determinate structure can experience thermal gradients between members with no loads incurred and no resistance to closure just prior to locking at the end of the deployment phase.

Table 4.3-4. Structural Materials Variation (LEO and GEO)

STRUCTURAL MATERIAL VARIATION*—20 POINTS		
CONCEPT	RANK	POINTS
1	1	20
2	1	20
3	6	8
4	1	20
5	6	8
6	1	20
7	1	20
8	6	8

NOTES:
*COMPOSITE MATERIALS OR ALUMINUM

Tables 4.3-5 and 4.3-6 compare, for GEO and LEO platforms, respectively, the candidate concept trusses accommodation of varied strength and stiffness requirements. The three ranges of strength and stiffness requirements are defined in Table 4.1-1, (Major Criteria used in the Selection Process). The accommodation of strength and stiffness is described by packaging efficiency and structure mass; hence, these parameters are the basis for this assessment. Clarification of each of the parameters is as follows:

o Packaging Efficiency

  The packaging efficiency is the ratio of deployed length to stowed length. This is a quantitative evaluation with linear distribution of points between the maximum and minimum values. Consideration was devoted to use of a volumetric efficiency term. The linear efficiency is used because the study of packaging (Section 1.4.3) the concepts indicates that it is the most significant factor.

o Mass Implication

  This parameter represents the estimated weight difference between the concepts as used in the generic platform. A detailed breakdown of the designs for the adopted strength and stiffness requirements is presented in Table 1.4-3. The concern here is the implication on GEO transfer cost.

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Table 4.3-5. Strength and Stiffness Accommodation (GEO)

CONCEPT	STRENGTH AND STIFFNESS ACCOMMODATION															
	REDUCED STRENGTH AND STIFFNESS					ADOPTED STRENGTH AND STIFFNESS					INCREASED STIFFNESS					TOTAL ACROSS ALL REQMTS.
	PACKAGING EFFICIENCY (PE)		MASS IMPLICATION			PACKAGING EFFICIENCY (PE)		MASS IMPLICATION			PACKAGING EFFICIENCY (PE)		MASS IMPLICATION			
	PE	PTS	Δ MASS	PTS		PE	PTS	Δ MASS	PTS		PE	PTS	Δ MASS	PTS		
	MAX POINTS 10		MAX POINTS 20		SUBTOTAL 30	MAX POINTS 20		MAX POINTS 30		SUBTOTAL 50	MAX POINTS 10		MAX POINTS 10		SUBTOTAL 20	100
1	22	5	1.0	12	17	15	10	1.1	17	27	8	5	10	0	5	49
2	32	10	1.2	10	20	21.6	17	1.3	15	32	8	5	14	0	5	57
3	26	7	0.0	20	27	20	15	0.0	30	45	20	9	0	10	19	91
4	26	7	0.2	19	26	20	15	0.7	22	37	10	6	10	0	6	69
5	25	6	0.9	13	19	25	20	0.1	29	49	25	10	3	3	13	81
6	31	9	0.6	15	24	20	15	1.0	18	33	10	6	12	0	6	63
7	25	6	1.3	10	16	25	20	1.3	15	35	25	10	3	3	13	64
8	28	8	0.6	15	23	22	17	0.9	18	35	14	7	12	0	7	65

NOTES: MASS DATA (KG x 10^{-3}) BASED ON GENERIC PLATFORM (WITHOUT MECHANIZATION SYSTEM).

Table 4.3-6. Strength and Stiffness Accommodation (LEO)

CONCEPT	STRENGTH AND STIFFNESS ACCOMMODATION						
	REDUCED STRENGTH AND STIFFNESS		ADOPTED STRENGTH AND STIFFNESS		INCREASED STIFFNESS		TOTAL ACROSS ALL REQMTS
	PACKAGING EFFICIENCY (PE)		PACKAGING EFFICIENCY (PE)		PACKAGING EFFICIENCY (PE)		
	PE	POINTS	PE	POINTS	PE	POINTS	
	MAXIMUM POINTS 10		MAXIMUM POINTS 20		MAXIMUM POINTS 10		40
1	22	5	15	10	8	5	20
2	32	10	21.6	17	8	5	32
3	26	7	20	15	20	9	31
4	26	7	20	15	10	6	28
5	25	6	25	20	25	10	36
6	31	9	20	15	10	6	30
7	25	6	25	20	25	10	36
8	28	8	22	17	14	7	32

Table 4.3-7 compares the candidate concepts' building-block platform construction versatility. The following discussion clarifies the compari//on parameters.

Table 4.3-7. Platform Construction (LEO and GEO)

PLATFORM CONSTRUCTION							
CONCEPT	EASE OF JOINING BUILDING BLOCKS		BEST ACCOMMODATION OF PAYLOADS AND DOCKING		MINIMUM DESIGN CONSTRAINTS— RAIL DEPLOYMENT		TOTAL
	RANK	POINTS	RANK	POINTS	RAIL LENGTH (METERS)	POINTS	
	MAX. POINTS 20		MAX. POINTS 20		MAX. POINTS 20		MAX. PTS 60
1	4	14	4	18	3.2	17	49
2	1	20	1	20	2.5	18	58
3	8	6	8	6	6.0	10	22
4	4	14	4	18	1.0	20	52
5	6	10	6	10	2.8	17	37
6	1	20	1	20	1.6	20	60
7	6	10	6	10	2.8	17	37
8	1	20	1	20	1.9	19	59

o Ease of Joining Building-Blocks

This section ranks the ease and versatility of joining building blocks together to form platforms of many configurations. Building blocks are joined to each other or to other modules via the main housing or adapter. Rigid square housings/adapters are the best, followed by rigid triangular shapes. Expanding triangular shapes, such as Concepts 5 and 7, pose difficulties because of the change in dimension of the housings/adapters. Concept 3 offers the most difficulty because of the flat shape of the housing/adapter, and because only two of the longerons are tied into the main housing. The other two longerons must be tied into another structure (subsequent to truss deployment) to maintain structural integrity.

o Best Accommodation of Payloads and Docking

Payloads/docking accommodations are provided by the main housings and the adapters. Adapters which do not change shape are judged superior to those which expand (Concepts 5 and 7) or to those which unfold (Concept 3). Rigid square main housings are best for mounting interfaces, closely followed by rigid triangular housings. Concepts 5 and 7 have expanding triangular housings which pose obvious difficulties. The mounting of an interface on the main housing of Concept 3 requires a deployable substructure.

- Minimum Design Constraints—Rail Deployment

  The length of the deployment/guide rails poses some design constraints. If the rail is to be folded for stowing, a longer rail may require multiple folds. If the rail is to be moved into operating position subsequent to a partial deployment of the truss/payload, a longer rail requires a longer "partial deployment" which in turn implies a longer "partial deployment mechanism". Finally, if a fixed rail system is to be used, a longer rail implies that more of the orbiter payload bay is required for stowage.

4.4 COSTS FOR GENERIC PLATFORM (LEO AND GEO)

The grading of the eight building-block concepts is shown on Table 4.4-1. Additional information is provided on Tables 1.4-4, and 1.4-5

The following notes are intended to explain the rationale behind the grading, points allocation of Tables 4.4-1.

Table 4.4-1 presents the comparative costs determined for the candidate concepts as used in the generic platform and sized for the adopted strength/stiffness requirements and adopted complement of utilities. The

Table 4.4-1. Costs for Generic Platform (LEO and GEO)

CONCEPT	Δ LAUNCH COST (1)		Δ FAB. COST	TOTAL Δ COST FOR LEO	EQUIV. POINTS FOR Δ COST FOR LEO	Δ OTV COST FOR GEO TRANSFER (2)		EQUIV. POINTS FOR Δ COST FOR GEO
	Δ PKG. LENGTH (METERS)	Δ ($M)	($M)	($M)	MAX PTS 20	Δ MASS	Δ ($M)	MAX PTS 20
1	2.25	5.9	1.5	7.4	11	1.1	9.7	12
2	1.5	3.9	4.3	8.2	10	1.3	11.5	10
3	2.3	6.0	0.3	6.3	12	0.0	0.0	20
4	0.75	1.8	0.5	2.3	17	0.7	6.2	14
5	0.0	0.0	0.0	0.0	20	0.0	0.0	20
6	2.1	5.5	2.3	7.8	11	1.0	8.8	12
7	0.0	0.0	1.3	1.3	18	1.3	11.5	10
8	1.5	3.9	3.0	6.9	12	1.2	10.6	11

NOTES:

(1) BASED ON $2.6M PER METER OF PACKAGED LENGTH

(2) BASED ON $8.8K PER KG

difference in packaged lengths is determined from the packaging arrangements shown in Section 1.4.3. The recurring fabrication costs are extracted from Table 1.4-5. The use of the unit Shuttle launch cost of $2.6M/meter is based upon a FY 1981 launch cost of $48M divided by the 18.3 m bay length. A significant reservation on the use of this value is that for a dedicated mission the $48M cost is incurred regardless of the length of the bay actually used.

The differences in platform mass are obtained from the data shown on Table 1.4-4. A significant reservation pertaining to the OTV transfer costs is that it is representative of only the adopted strength/stiffness requirements and for the generic platform. For this reason, and the reservations noted above, the point allotment of the cost criteria on the final summation charts is no more than 40. Further, the maximum cost differences shown are very small compared to the total system cost.

4.5 THERMAL STABILITY OF STRUCTURAL CONCEPT

The grading of the eight concepts for thermal stability is shown in Table 4.5-1. This table is a summary of the thermal data shown on Table 1.4-2.

Table 4.5-1: Thermal Stability of Structural Concept

CONCEPT	d (m)	DIAGONAL PRESENT	LONGERON TEMP. (°C)		ΔT (°C)	FIGURE OF MERIT, d/(ΔT)	POINTS
			SUN SIDE	SHADE SIDE			
1	1.6	YES	21.6	-13.5	35.1	0.046	11
2	1.3	YES NO	20.7 24.0	-11.6 4.8	32.3 19.2	0.053	12
3	3.0	NO	24.0	2.7	21.3	0.140	20
4	1.7	YES	21.7	-7.4	29.1	0.058	12
5	2.5	YES (TENSION STRAP)	23.8	-36.0	59.8	0.040	11
6	1.3	YES	22.5	-22.6	45.1	0.029	10
7	2.5	YES	23.6	-56.7	80.3	0.031	10
8	1.3	YES	22.5	-17.5	40.0	0.031	10

4.6 METEOROID IMPACT SUITABILITY OF STRUCTURAL CONCEPT

This section discusses the issue of potential meteoroid impact and structural survival. The data in Table 4.6-1 were derived using the meteoroid model stipulated in Reference 6. The model is sufficiently accurate for the GEO environment. Man-made debris was less critical (Reference 15).

The size of the meteoroid particles and associated probabilities are shown for two sizes of platform and for a 10-year exposure. The projected area applies to the totality of truss members. Little to no recent information exists pertinent to the size of holes resulting from the meteoroid strike, particularly for graphite composites. From discussions with Materials personnel and reviews of meteoroid impact damage (Reference 7), it is estimated that the hole size may be 2 to 4 times the diameter of the particle. Considering the low levels of stress and low number of cycles associated

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with RCS systems, and the negligible impact of the hole on local or Euler stability, it is likely that the structural damage will be acceptable. However, since this needs to be verified, redundancy in the structural design is an advantage; hence the grading of the eight concepts is based on that consideration and is shown in Table 4.6-2.

Table 4.6-1. Probability of Meteoroid Damage

GENERIC PLATFORM (220 m^2)		
PROBABILITY OF HIT IN 10 YR (%)	METEOROID DIAMETER (cm)	POTENTIAL HOLE DIA. (cm)
1	0.60	1-1/4 TO 2-1/2
2	0.48	1 TO 2
5	0.38	3/4 TO 1-1/2
SMALLER PLATFORM (70 m^2)		
PROBABILITY OF HIT IN 10 YR (%)	METEOROID DIAMETER (cm)	POTENTIAL HOLE DIA. (cm)
1	0.42	3/4 TO 1-1/2
2	0.36	3/4 TO 1-1/2
5	0.30	1/2 TO 1-1/4

Table 4.6-2. Meteoroid Impact Suitability

CONCEPT	METEOROID IMPACT SUITABILITY—20 POINTS	
	RANK	POINTS
1	5	10
2	4	16
3	1	20
4	5	10
5	5	10
6	1	20
7	5	10
8	1	20

4.7 RELIABILITY OF DEPLOYMENT OF BUILDING BLOCK

This section compares the candidate building-blocks for reliability of deployment based on the parameters which are explained in detail below. The grading of the eight concepts is listed in Table 4.7-1.

- o Basic Truss Based on Number of Joints.

This evaluation is based on the number of joints in the length of truss required to build the generic platform. The joints included are sliding joints in diagonals and battens, and folding/rotating joints in the longerons and pyramidal members. There is an inverse linear relationship between the number of joints and the number of points awarded. Table 4.7-2 describes in detail the numbers of joints for each of the eight concepts.

- o Basic Truss Based on Complexity.

This is an assessment based on the type of joints/ movements used in deploying the basic truss structure. Sliding joints in diagonals are judged to be more complex than folding joints. The I-section sliding battens of Concepts 5 and 7 are judged to be the most complex.

Table 4.7-1. Reliability of Building-Block Deployment

CONCEPT	BASIC TRUSS BASED ON NUMBER OF JOINTS		BASIC TRUSS BASED ON COMPLEXITY		THERMAL EFFECTS— GRAPHITE COMPOSITE TRUSS		THERMAL EFFECTS— ALUMINUM TRUSS		HOUSING STRUCTURE		ADAPTER STRUCTURE		DOCKING PORT SUPPORT STRUCTURE		TOTAL
	NO. OF JOINTS	PTS	RANK	PTS	RANK	PTS	RANK	PTS	RANK	PTS	RANK	PTS	RANK	PTS	
	MAX POINTS 10		MAX POINTS 10		MAX POINTS 15		MAX POINTS 5		MAX POINTS 20		MAX POINTS 20		MAX POINTS 10		MAX POINTS 90
1	848	9	2	8	1	15	1	5	1	20	1	20	1	10	87
2	1350	8	2	8	6	12	6	4	1	20	1	20	1	10	82
3	1368	8	1	10	7	12	7	3	1	20	8	12	8	4	69
4	1242	8	2	8	1	15	1	5	1	20	1	20	1	10	86
5	732	10	6	5	7	12	7	3	7	8	6	16	6	6	60
6	1660	7	2	8	1	15	1	5	1	20	1	20	1	10	85
7	1464	8	6	5	1	15	1	5	7	8	6	16	6	6	63
8	2208	5	6	5	1	15	1	5	1	20	1	20	1	10	80

Table 4.7-2. Number of Joints in Basic Structure for Generic Platform

CONCEPT	NUMBER OF BAYS	NUMBER OF LONGERON FOLDED JOINTS	NUMBER OF DIAGONAL TELESCOPING JOINTS	NUMBER OF DIAGONAL FOLDED JOINTS	NUMBER OF BATTEN TELESCOPING JOINTS	NUMBER OF PYRAMIDAL JOINTS	TOTAL JOINTS
1	212	636	212	—	—	—	848
2	270	1080	270	—	—	—	1350
3	114	456	—	—	—	912	1368
4	207	621	621	—	—	—	1242
5	122	366	—	—	366	—	732
6	215	830	830	—	—	—	1660
7	122	366	366	732	366	—	1464
8	184	1472	736	—	—	—	2208

NOTES: COUNT DOES NOT INCLUDE BASIC CLEVIS JOINTS BECAUSE THEY ARE SIGNIFICANTLY LESS COMPLEX THAN FOLDING AND TELESCOPING JOINTS. THE NUMBERS ARE BASED ON APPROXIMATELY 340 METERS OF TRUSS, WHICH IS THE LENGTH REQUIRED TO BUILD THE GENERIC PLATFORM.

- Thermal Effects, Graphite Composite Truss or Aluminum Truss

These parameters account for the reduced reliability of deployment inherent in the structures that are indeterminate for the materials shown. The values are judgmental between the maximums and minimum shown.

- Housing Structure

Concepts 1, 2, 3, 4, 6, and 8 are ranked equal because they are rigid with no mechanisms required. Concepts 5 and 7 are ranked last, because they require a mechanism for lateral extension.

- Adapter Structure

Concepts 1, 2, 4, 6, and 8 are ranked first because they are rigid structures which require no mechanisms. Concepts 5 and 7 are judged as having less reliability because they are expanded by a mechanism. Concept 3 is ranked last because it requires a separate mechanism to unfold it.

- Docking Port Structure

A docking port interface mounted to a rigid main housing is the most reliable. Concepts 5 and 7 have expanding main housings which degrade the reliability of the interface. Concept 3 requires a separate substructure to be deployed to obtain a docking port interface. This requires additional mechanization, which is the reason for its being ranked 8th.

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4.8 PREDICTABILITY OF PERFORMANCE OF STRUCTURAL CONCEPTS

The grading for the eight concepts is listed in Table 4.8-1.

The features of the eight structural concepts which affect the accurate prediction of structural performance are:

- A determinate structure is better than an indeterminate structure for analytical purposes
- The difficulty of maintaining the tension in "X" braced structures which is essential to performance predictability
- The disadvantages of designs with offset load paths at the joints

While NASTRAN analysis and development testing during a program can deal with these effects, Table 4.8-1 judgmentally and qualitatively accounts for these additional requirements.

Table 4.8-1. Predictability of Performance

CONCEPT	COMPOSITE MATERIALS: PREDICTION OF INTERNAL LOADS — RANK	POINTS (MAX POINTS 10)	PREDICTION OF EFFECTIVE STIFFNESS — RANK	POINTS (MAX POINTS 10)	TOTAL (MAX PTS 20)	ALUMINUM: PREDICTION OF INTERNAL LOADS — RANK	POINTS (MAX POINTS 10)	PREDICTION OF EFFECTIVE STIFFNESS — RANK	POINTS (MAX POINTS 10)	TOTAL (MAX PTS 20)	TOTAL FOR BOTH MATERIALS
1	1	10	1	10	20	1	10	1	10	20	40
2	6	9	8	5	14	6	9	7	5	14	28
3	8	6	1	10	16	7	4	1	10	14	30
4	1	10	1	10	20	1	10	1	10	20	40
5	6	6	7	8	14	8	4	8	2	6	20
6	1	10	1	10	20	1	10	1	10	20	40
7	1	10	1	10	20	1	10	1	10	20	40
8	1	10	1	10	20	1	6	1	10	16	36

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4.9 ORBITER INTEGRATION SUITABILITY

This section compares the candidate designs in regard to their orbiter integration suitability.

The grading of the eight concepts is shown on Table 4.9-1.

Table 4.9-1. Orbiter Integration Suitability

CONCEPT	③ HOUSING LAUNCH ENVIRONMENT SUITABILITY		① EASE OF PACKAGING INTO ORBITER		② EASE OF PACKAGING INTO ORBITER		① CRADLE STRUCTURE Δ MASS (KG x 10^{-3})		① COMPLEXITY, PACKAGED CONFIGURATION TO DEPLOYED		SPARE VOLUME		
	RANK	POINTS	RANK	POINTS	RANK	POINTS	Δ MASS	POINTS	RANK	POINTS	RANK	POINTS	TOTAL
	MAX POINTS 10		MAX POINTS 10		MAX POINTS 10		MAX POINTS 10		MAX POINTS 10		MAX POINTS 10		MAX PTS 60
1	1	10	8	5	1	10	0.9	8	3	8	7	5	46
2	1	10	5	6	1	10	1.4	7	5	7	4	7	47
3	8	3	4	7	1	10	2.3	5	8	6	1	10	41
4	1	10	3	8	1	10	1.0	8	3	8	7	5	49
5	1	10	1	10	6	8	0	10	1	10	2	8	56
6	1	10	5	6	1	10	1.6	6	5	7	4	7	46
7	1	10	1	10	6	8	0	10	1	10	2	8	56
8	1	10	5	6	1	10	1.4	7	5	7	4	7	47

NOTES: ① APPLICABLE TO GENERIC PLATFORM
② APPLICABLE TO PLATFORM SMALLER THAN GENERIC
③ BASIC BUILDING BLOCK CONCEPT

A discussion of each of the design parameters, listed above, is as follows:

- Housing Launch Environment Suitability

This parameter judgmentally accounts for the relative capability of the candidate housing designs to sustain the launch/inertia induced loads and also to provide (in conjunction with the cradle structure) a minimum natural frequency above that of the orbiter (say, 10 Hz).

- Ease of Packaging into Orbiter (Generic Platforms)

The rankings of this section are based on the studies and drawings made for the generic platform only. Factors which influence the package are:

- Packaging ratio = $\dfrac{\text{expanded length}}{\text{stowed length}}$ of a truss
- The shape of the truss section
- The size of the truss section

Concepts 5 and 7 gain first due to their high packaging ratio. Triangular-shaped trusses generally fit together in a circle (such as the orbiter payload bay) better than square-shaped trusses. If the packaging can be arranged so that the building blocks fit across the payload bay instead of along the longitudinal axis, it is an advantage. This is reflected in the ranking of Concept 3.

- Ease of Packaging into Orbiter (Smaller than Generic)

No drawings of packaging "smaller than generic" platforms into the orbiter were made. As the platforms become smaller, so does the differential between them, until the point is reached for <u>very</u> small platforms when there is generally little significant advantage of one concept over another. It is judged that small platforms can probably be packaged across the width of the payload bay. Concepts 5 and 7 are ranked slightly lower because of the need for clearance around the expanding main housings.

- Cradle Structure Delta Weight.

This quantitative assessment describes the delta weight additional to each concept as packaged in the orbiter. The delta weight represents cradle structure, trunnions and, where required, reinforcement of the housing structure. No distinction is made between LEO and GEO designs at this stage of the investigation.

- Complexity of Configuration—Packaged to Deployed

This evaluation addresses the difficulty and complexity of moving the building blocks from the packaged configuration to the final deployed configurations. For each concept, a count was made of the number of mechanisms/movements required to unlock, rotate, relock each building-block housing and adapter, and the unfolding of the deployment/guide rails. The concept with the least number of mechanisms/movements, etc., was awarded first place with the other concepts graded accordingly. This assessment is applicable only to the generic platform.

- Spare Volume

In the length of the orbiter payload bay which is occupied by the packaged generic platform, a certain percentage of the volume is available for other purposes. The concept with the maximum space is awarded first place, and the other concepts are ranked accordingly. This assessment applies only to the generic platform.

4.10 SUMMARY OF POINTS AND GRADING

The results of the points allocation and grading of Sections 4.3 through 4.9 are presented in five tables:

- o Table 4.10-1, Total of Normalized Points (LEO)
- o Table 4.10-2, Total of Normalized Points (GEO)
- o Table 4.10-3, Total of Normalized Points (LEO)-Sensitivity Trade
- o Table 4.10-4, Total of Normalized Points (GEO)-Sensitivity Trade
- o Table 4.10-5, Summary of Points (LEO and GEO)

Table 4.10-1 (LEO) presents the total points in each of the seven major selection criteria for each building-block concept. The point values shown for each concept are obtained from the total point value determined in the individual preceding criteria sheets multiplied by the appropriate factor to be compatible with the maximum point allocations shown on this chart for each criterion. For example, for Concept 1, the total points for "Reliability of Building-Block Deployment" (obtained from Table 4.7-1) is 87 out of a total possible 90 points. The 97 points shown on Table 4.10-1 is obtained by 100/90 x 87 = 96.66 or 97. The "Design Versatility" critorion includes the points obtained from Tables 4.3-2, 4.3-3, 4.3-4, 4.3-6 and 4.3-7, normalized in the same manner as shown above.

Table 4.10-1. Total of Normalized Points (LEO)

CONCEPT	(1) DESIGN VERSATILITY	(2) COST	(3) THERMAL STABILITY	(4) METEOROID IMPACT SUITABILITY	(5) RELI-ABILITY	(6) PREDICT-ABILITY	(7) ORBITER INTEGRATION	TOTAL
	MAX POINTS 100	MAX POINTS 40	MAX POINTS 20	MAX POINTS 40	MAX POINTS 100	MAX POINTS 20	MAX POINTS 60	MAX POINTS 380
1	87 ✓	22	11	20	97 ✓	20 ✓	46	303
2	94 ✓	20	12	32	91	14	47 ✓	310 ③
3	60	24	20 ✓	40 ✓	77	15	41	277
4	78	34 ✓	12	20	96 ✓	20 ✓	49	309 ④
5	80	40 ✓	11	20	67	10	56 ✓	284
6	83	22	10	40 ✓	94 ✓	20 ✓	46	315 ②
7	84	36 ✓	10	20	70	20 ✓	56 ✓	296
8	88 ✓	24	10	40 ✓	89	18	47 ✓	316 ①

NOTES:

CIRCLED NUMBERS IN TOTALS COLUMN DENOTE RANKING.

✓ FOR TOP 3 VALUES IN EACH CATEGORY

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The assignment of maximum points of 100 to "design versatility" and "reliability" represents the greater importance of these criteria. The rationale for reduced emphasis on cost is discussed in Section 4.4. It is important to note that this generic requirements point allocation would be different for a very specific mission, depending on constraints. For example, suppose the available orbiter length is a constraint, or suppose subsequent test data indicate meteoroid impact to be more critical than expected. These parameters and/or criteria take on an increased significance. The total points are shown to the right for each concept, with Concepts 8, 6, 2, and 4 representing the most suitable design for a LEO platform.

Table 4.10-2 (GEO) is compiled in the same fashion as Table 4.10-1. The table indicates Concepts 6, 8, 4, 3, and 2 to be most suitable.

Table 4.10-2. Total of Normalized Points (GEO)

CONCEPT	(1) DESIGN VERSATILITY	(2) COST	(3) THERMAL STABILITY	(4) METEOROID IMPACT SUITABILITY	(5) RELI-ABILITY	(6) PREDICT-ABILITY	(7) ORBITER INTEGRATION	TOTAL
	MAX POINTS 100	MAX POINTS 40	MAX POINTS 20	MAX POINTS 40	MAX POINTS 100	MAX POINTS 20	MAX POINTS 60	MAX POINTS 380
1	77	24	11	20	97 ✓	20 ✓	46	295
2	86 ✓	20	12	32	91	14	47 ✓	302 ④
3	69	40 ✓	20 ✓	40 ✓	77	15	41	302 ④
4	78	28 ✓	12	20	96 ✓	20 ✓	49	303 ③
5	75	40 ✓	11	20	67	10	56 ✓	279
6	80 ✓	24	10	40 ✓	94 ✓	20 ✓	46	314 ①
7	75	20	10	20	70	20 ✓	56 ✓	271
8	81 ✓	22	10	40 ✓	89	18	47 ✓	307 ②

NOTES:
CIRCLED NUMBERS IN TOTALS COLUMN DENOTE RANKING.
✓ FOR TOP 3 VALUES IN EACH CATEGORY

Tables 4.10-3 and 4 describe the corresponding sensitivity data derived as described in Section 4.2. The results are summarized in Table 4.10-5. The sensitivity study data, however, is not considered on an equal basis with that of the baseline.

Table 4.10-3. Total of Normalized Points (LEO)—Sensitivity Trade

CONCEPT	(1) DESIGN VERSATILITY MAX POINTS 100	(2) COST MAX POINTS 40	(3) THERMAL STABILITY MAX POINTS 20	(4) METEOROID IMPACT SUITABILITY MAX POINTS 40	(5) RELI-ABILITY MAX POINTS 100	(6) PREDICT-ABILITY MAX POINTS 20	(7) ORBITER INTEGRATION MAX POINTS 60	TOTAL MAX POINTS 380
1	75	4	2	20	93 ✓	20 ✓	44	258
2	90 ✓	0	4	32	85	14	44	269
3	53	8	20 ✓	40 ✓	71	15	36	243
4	70	28 ✓	5	20	91 ✓	20 ✓	47 ✓	281 (1)
5	76	40 ✓	1	20	62	10	56 ✓	265
6	79	0	0	40 ✓	89 ✓	20 ✓	43	271 (4)
7	80 ✓	34 ✓	0	20	64	20 ✓	56 ✓	274 (2)
8	82 ✓	6	0	40 ✓	83	18	44	273 (3)

NOTES:
Circled numbers in totals column denote ranking.
✓ For top 3 values in each category.

Table 4.10-4. Total of Normalized Points (GEO)—Sensitivity Trade

CONCEPT	(1) DESIGN VERSATILITY MAX POINTS 100	(2) COST MAX POINTS 40	(3) THERMAL STABILITY MAX POINTS 20	(4) METEOROID IMPACT SUITABILITY MAX POINTS 40	(5) RELI-ABILITY MAX POINTS 100	(6) PREDICT-ABILITY MAX POINTS 20	(7) ORBITER INTEGRATION MAX POINTS 60	TOTAL MAX POINTS 380
1	58	6	2	20	93 ✓	20 ✓	44	243
2	76 ✓	0	4	32	85	14	44	255
3	65	40 ✓	20 ✓	40 ✓	71	15	36	287 (1)
4	69 ✓	18 ✓	5	20	91 ✓	20 ✓	47 ✓	270 (3)
5	66	40 ✓	1	20	62	10	56 ✓	255
6	73 ✓	10	0	40 ✓	89 ✓	20 ✓	43	275 (2)
7	61	0	0	20	64	20 ✓	56 ✓	221
8	69 ✓	4	0	40 ✓	83	18	44	258 (4)

NOTES:
Circled numbers in totals column denote ranking.
✓ For top 3 values in each category.

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Table 4.10-5 summarizes the data obtained from the preceding tables. The number of criteria for each concept that are within the top three is indicated.

The major result is that there is little difference in the total points obtained for the top four designs. Hence, the decision between these is judgemental, but based on the knowledge gained from the details of the study and tabulation of the selection process data.

Table 4.10-5. Summary of Points and Grading, LEO and GEO

	LEO PLATFORM						GEO PLATFORM					
	BASELINE DATA		SENSITIVITY DATA		NO. OF CRITERIA IN TOP 3		BASELINE DATA		SENSITIVITY DATA		NO. OF CRITERIA IN TOP 3	
CONCEPT	PLACE	POINTS	PLACE	POINTS	BASE	SENS.	PLACE	POINTS	PLACE	POINTS	BASE	SENS.
8	1	316	3	273	3	2	2	307	4	258	3	2
6	2	315	4	271	3	3	1	314	2	275	4	4
4	4	309	1	281	3	4	3	303	3	270	3	5
2	3	310	5	269	2	1	4	302	5	255	2	1

4.11 CONCEPT SELECTION

A review of the selection process of Sections 4.1 through 4.10 and Table 4.10-5 leads to the following general conclusions:

- Concept selection between the first four designs is judgmental, but based on knowledge obtained from selection process study.
- Specific mission critical requirements will impose special emphasis on particular parameters and concept selection.
- No clear concept choice distinction for LEO or GEO application is represented by the data.
- The major difference in concept selection will result from extent of payload power and data requirements.
- No one design is best on the basis of satisfaction of every requirement - improvement toward one requirement almost always results in degradation of another requirement.
- The thrust of the program for the next 4 years should be to resolve the major design & technology issues pertinent to deployable platform systems.

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- o The major design and technology issues are essentially the same across the candidate designs.

- o The open-square shape of Concept 8 has best growth potential for utilities since it can accommodate in trays up to twice the baseline study requirements (increased utilities requirements typical with program maturity).

- o For significantly reduced utilities requirements (mountable on longerons), Concept 6 is adequate and is a simpler structural design than Concept 8.

- o Concepts 8 and 6 together present the most promising combinations of top four designs (Figure 4.11-1).

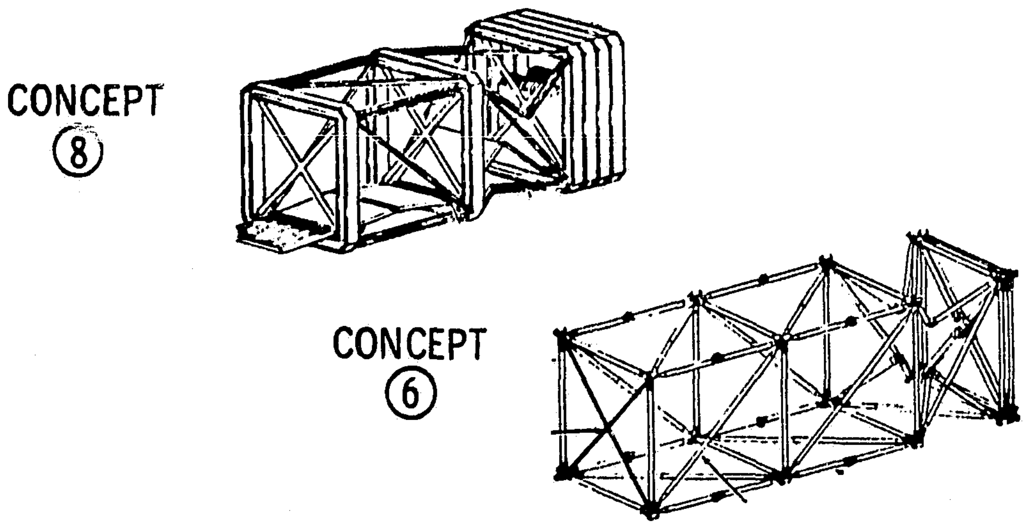

Figure 4.11-1. Concepts 6 and 8

Subsequent to the identification of Concepts 6 and 8 as the two designs best suited for LEO/GEO platforms, it became evident that the best features of both can be combined into one. This concept, known as 6A, is shown in Figure 4.11-2, and further discussed as follows:

- o Concept 6A is the best utilization of the advantages of Concept 8 (utilities accommodation) and Concept 6 (structural simplicity).

- o Concept 6A permits full accommodation of the adopted utilities requirement with a 0.5-meter-wide tray. The longerons are folded at 30°.

- o Concept 6A has an acceptable packaging efficiency of 18 for the adopted requirements, and a 1.26-m-deep truss. Increased packaging efficiency is achievable with increased truss depth and reduced loads.

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- The technology development of Concept 6A is applicable to a family of designs ranging from Concept 6A to Concept 6.

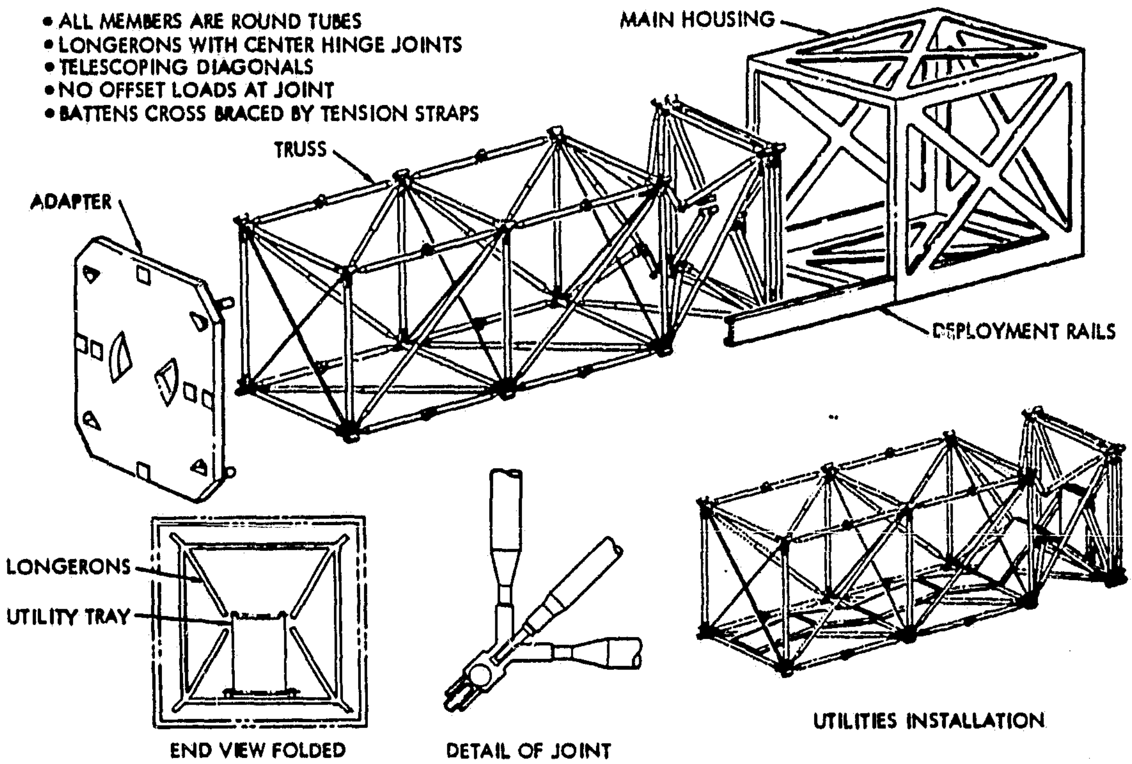

Figure 4.11-2. Concept 6A

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5. DEPLOYABLE VOLUME ENCLOSURES

This section describes several applications of deployable volumes for a typical Space Station (Figure 5.0-1). The three specific applications are habitat modules, an orbit transfer vehicle (OTV) hangar, and crew transfer tunnels.

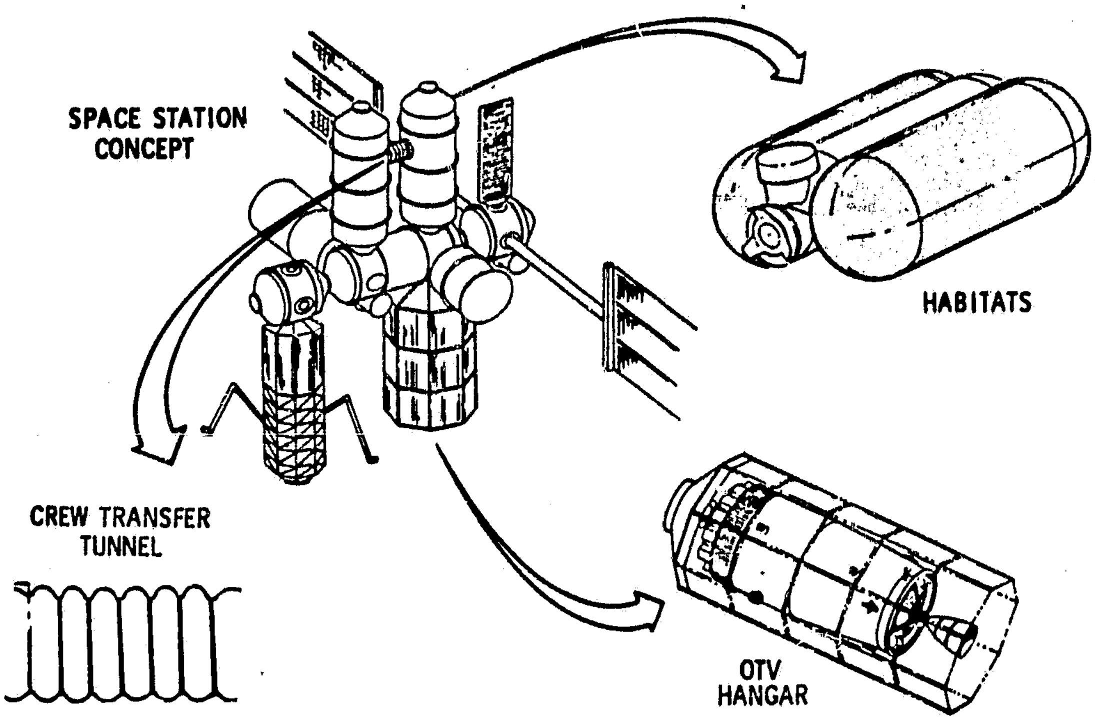

Figure 5.0-1. Deployable Volume Enclosures

Current designs for Space Station modules typically visualize rigid bodies which are approximately the same size as the orbiter payload bay. With the advent of deployable volumes there arises the possiblity of launching much larger modules without the problems associated with assembly in space. This section examines these possibilities and demonstrates that in many cases they are more than just possibilities - they are emimently feasible.

5.1 HABITAT MODULES

5.1.1 Habitat Requirements

The requirements for the design of the habitat module are:

- o A life of 10 to 20 years in LEO.
- o Compatiblity with the orbiter for transportation to LEO.
- o Compatiblity with existing designs and concepts for manned space stations.

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- o Accommodation of crew of up to 8 for 90 days + 90 days emergency.
- o Normal pressurization of 14.7 psi (limit); 8 psi for an emergency.
- o Provision of two separate pressurized compartments.
- o Provision of two routes for ingress/egress.
- o Automatic deployment to maximum extent feasible, EVA assist if advantageous.
- o Incorporation of all the equipment normally associated with habitat modules, including life support equipment, command control center etc.
- o Provision of two docking systems.
- o Minimum of one airlock.
- o Exterior mounting of equipment such as fuel cells, toxic items etc.
- o Radiation protection of 0.50 gram/cm^2 (1.0 lb/ft^2).
- o Adequate meteoroid protection (discussed subsequently).

5.1.2 Radiation/Meteoroid Shielding Review

The suitability of a potential habitat wall design concept (gratuitously furnished by Goodyear Aircraft Corp.) for radiation shielding has been evaluated (Figure 5.1-1). The basic 2.5-cm-thick design with a foam density of 0.032 gm/cc^3 is adequate for precluding skin damage, but not for protection of the astronauts' eyes. An additional 0.24 gm/cc^2 needs to be provided by either increasing the bladder thickness, foam thickness or density, a combination of the foregoing, or other means. The implication on packaging needs to be assessed (in subsequent studies).

The associated probabilities of meteoroid puncture of the inner wall using two different foam densities are shown. These data are based upon the meteoroid model specified in Reference 6 and the man-made debris model of Reference 15. The analyses assume an effective stopping power of 15 for the foam, i.e., the foam is as effective as 15 times the same mass per unit area of a sheet of aluminum. This information was obtained from documented tests performed by Goodyear, and is consistent with predictions by Rockwell researchers.

The probabilities shown indicate that with development of a leak-detection system and repair capability, the inflatable wall design shown can be suitable insofar as meteoroid impact is concerned (propagation of the puncture is not expected to occur).

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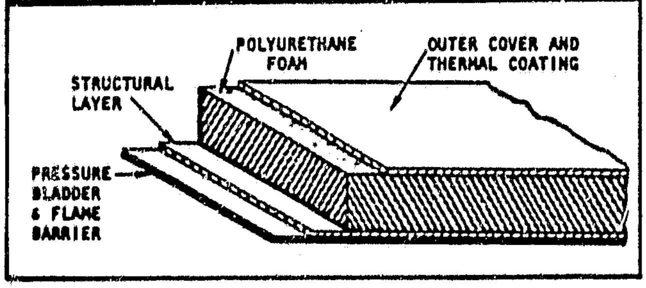
ELASTIC RECOVERY CONCEPT*



FOR PROJECTED AREA OF 100 m^2

0.016 GM/CM³ FOAM (2.5 cm THICK)

- 1% PROBABILITY OF PUNCTURE IN 3 DAYS
- ON AVERAGE, ONE PUNCTURE EVERY 9 MONTHS TO BE REPAIRED

0.032 GM/CM³ FOAM (2.5 cm THICK)

- 1% PROBABILITY OF PUNCTURE IN 27 DAYS
- ON AVERAGE, ONE PUNCTURE IN 7.5 YEARS TO BE REPAIRED

DETAIL WEIGHTS OF EXPANDABLE MATERIALS*

CONSTRUCTION	(gm/cm^2)
• ALUMINUM INNER LAYER	0.002
• ADHESIVE	0.005
• PRESSURE BLADDER	0.080
• ADHESIVE	0.005
• STRUCTURAL LAYER	0.031
• TASLAN INTERLOCKING LAYER AND ADHESIVE	0.024
• POLYURETHANE FOAM	0.080
• ADHESIVE	0.005
• OUTER COVER AND COATING	0.031
TOTAL	$\boxed{0.263}$

ADDITIONAL MATERIAL REQUIRED TO SATISFY RADIATION REQMT.

*Courtesy of Goodyear Aerospace (for Expandable Airlock Technology Experiment)

Figure 5.1-1. Habitat Radiation/Meteoroid Implications

5.1.3 Habitat Design Approach

A review of the habitat requirements and logistics associated with delivery of habitats to LEO resulted in the following design approach.

- The habitat module should be large as compared to conventional modules to significantly offset the reduced cost and higher reliability of a conventional design.
- For habitats with inflatables, use combination of hard (fixed) structure and inflatables.
- Design to accommodate equipment in its correct locations (on the hard structure) during Shuttle launch to minimize work/ rearrangement on orbit.
- Separate the crew quarters from the equipment not in regular use with placement of heavy equipment on hard structure and crew quarters in the deployable structure.
- Divide the crew quarters into large volumes for communal use and into smaller private volumes.
- Build radiators into the exterior of the hard structure.
- Provide capability to repair inflatables from the inside.
- A meteoroid bumper is desirable for inflatables.

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- Utilize the crew inside the Habitat Module for relocation of minor equipment, final stages of deployment such as locking joints and removal of temporary deployment mechanisms.

5.1.4 Candidate concepts for Habitat Modules

Eight concepts for deployable Habitat Modules were considered (Figure 5.1-2). Three were selected for further study (Section 5.1.5).

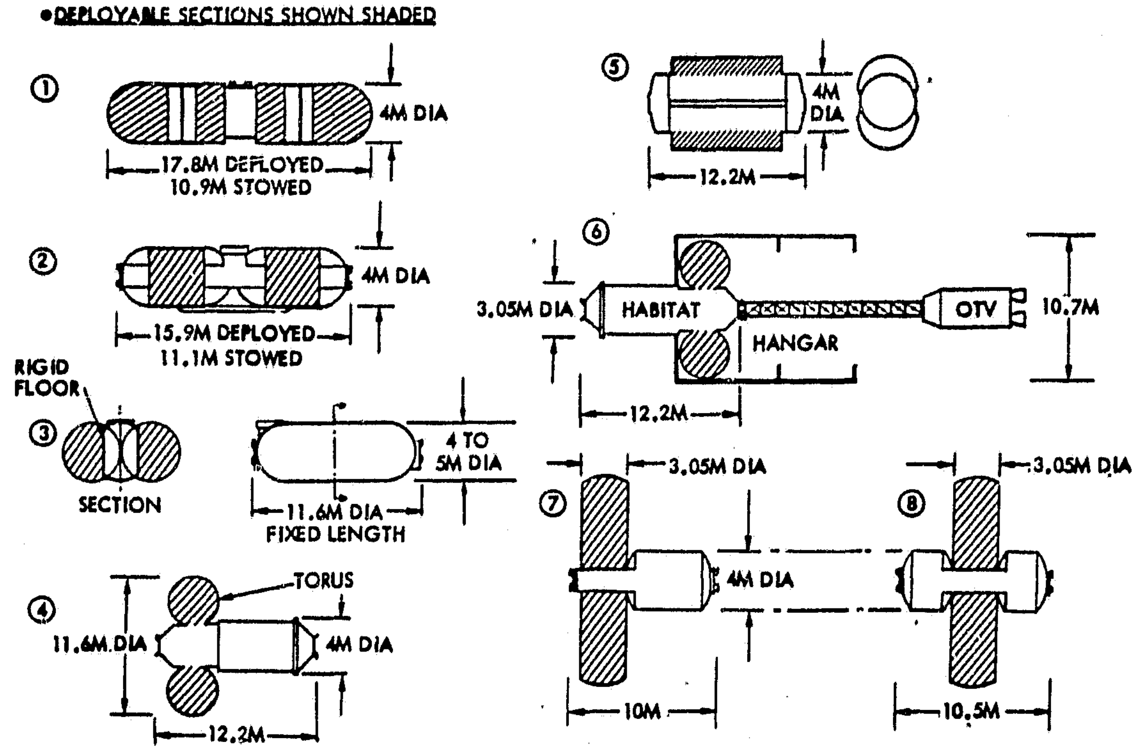

Figure 5.1-2. Candidate Deployable Habitat Module Concepts

Concept 1 expands axially from a central airlock/docking port and two hard decks on which equipment is mounted. The stowed module occupies most of the orbiter bay and the deployed volume is minimal. The inflatable sections are supported by light deployable internal structures.

Concept 2 is a derivative of Concept 1 but has spherical end bulkheads.

Concept 3 has a rigid central floor which acts also as a strongback for launch. There are two docking ports and an airlock. The two inflatable sections are shown as approximately 4.5 meters in diameter but can be larger. They can be subdivided into wardroom, sleeping quarters, etc., as desired.

Concept 4 consists of a nearly "standard" rigid habitat module with an inflatable torus attached. It has excellent capability for mounting equipment, good radiator area and a large inflatable volume for crew quarters. The arrangement for airlocks and docking is the same as for "standard" rigid habitat modules.

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Concept 5 is a "standard" rigid habitat module with two deployable hard sections. The deployable sections can be sealed either by large internal bags or by sealing around individual joints (or both). The volumes available for equipment and crew quarters are excellent. This concept has the further advantages of all-hard structure, large radiator area, standard docking, and no unusual structures.

Concept 6 consists of Concept 4 with the addition of a hard deployable shell which serves as a meteoroid bumper for the habitat module and as an OTV hanger. The system is deployable and is packaged for a single launch.

Concept 7 is a rigid module with two inflatable sections deployed from it. It is somewhat similar to Concept 4 but the deployed volume is probably not as useful.

Concept 8 is a variation of Concept 7, as shown.

5.1.5 Preferred Concepts

The three preferred concepts are shown in Figure 5.1-3. The drawings are contained in Volume II. These concepts were selected for further investigation for the following reasons;

- o Representation of a wide variety of designs
- o A good ratio of deployed volume/stowed volume is provided

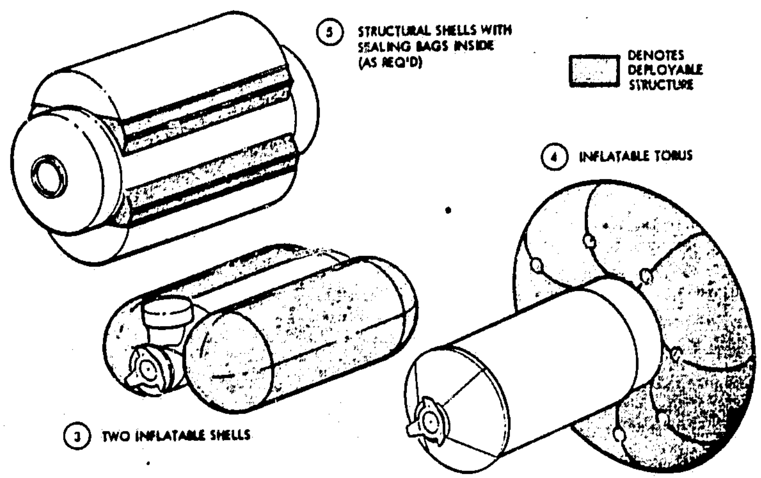

Figure 5.1-3. Preferred Habitat Module Concepts

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- o High growth potential
- o Adequate hard structure for mounting equipment
- o Safety (redundant volumes available)
- o The crew quarters can be subdivided into large and small rooms as desired
- o No technology development of an unusually difficult nature is required

5.1.5.1 Habitat Concept 3

Concept 3 (Figure 5.1-4) consists essentially of an inflatable shell mounted on each side of a strongback. The strongback serves as a launch cradle, (contains trunnions and keel fitting), a mounting platform for equipment, and as a structural support for the airlocks and docking systems.

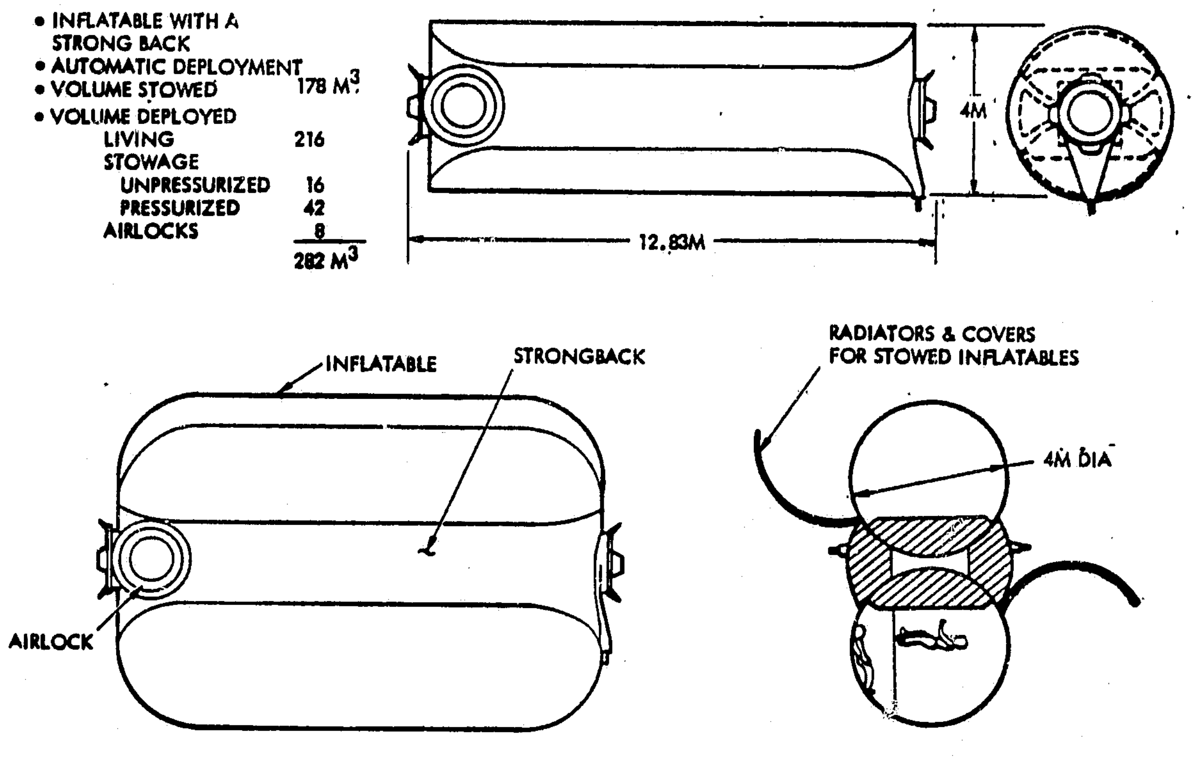

Figure 5.1-4. Habitat Module (Concept 3)

The inflatable shells are shown as 4.5 meters diameter but they can be larger. When stowed for launch, they are contained behind covers which conform to the shape of the orbiter payload bay. The covers are deployed to allow the shells to inflate and are then used as radiators. The deployed module forms two separate sections for safety/redundancy, with the strongback providing a long central floor. Equipment is mounted both inside and outside of the pressurized volumes. The two docking systems conform to standard habitat module practice for design and placement.

5.1.5.2 Habitat Concept 4

This design (Figure 5.1-5) is a combination of an inflatable section and a rigid body. When stowed, the module occupies all of the orbiter payload bay with the exception of space for the docking module and the OMS kit. The module is supported in the payload bay by standard trunnions and a keel fitting and requires no separate cradle.

The rigid body is used mainly for equipment, arranged around a central tunnel which runs the length of the module. The outside diameter of the rigid body is used as a radiator for the whole of the Habitat Module. The inflatable section is in the form of a torus which surrounds one end of the rigid body. The torus provides a large living volume which can be used in many ways. One requirement is probably to divide it into two spaces for safety/redundancy in the event of loss of pressure. It should be noted that there is space inside the rigid body which can be used in an emergency. A docking mechanism is provided at one end of the module while at the other end an airlock with a docking mechanism is provided.

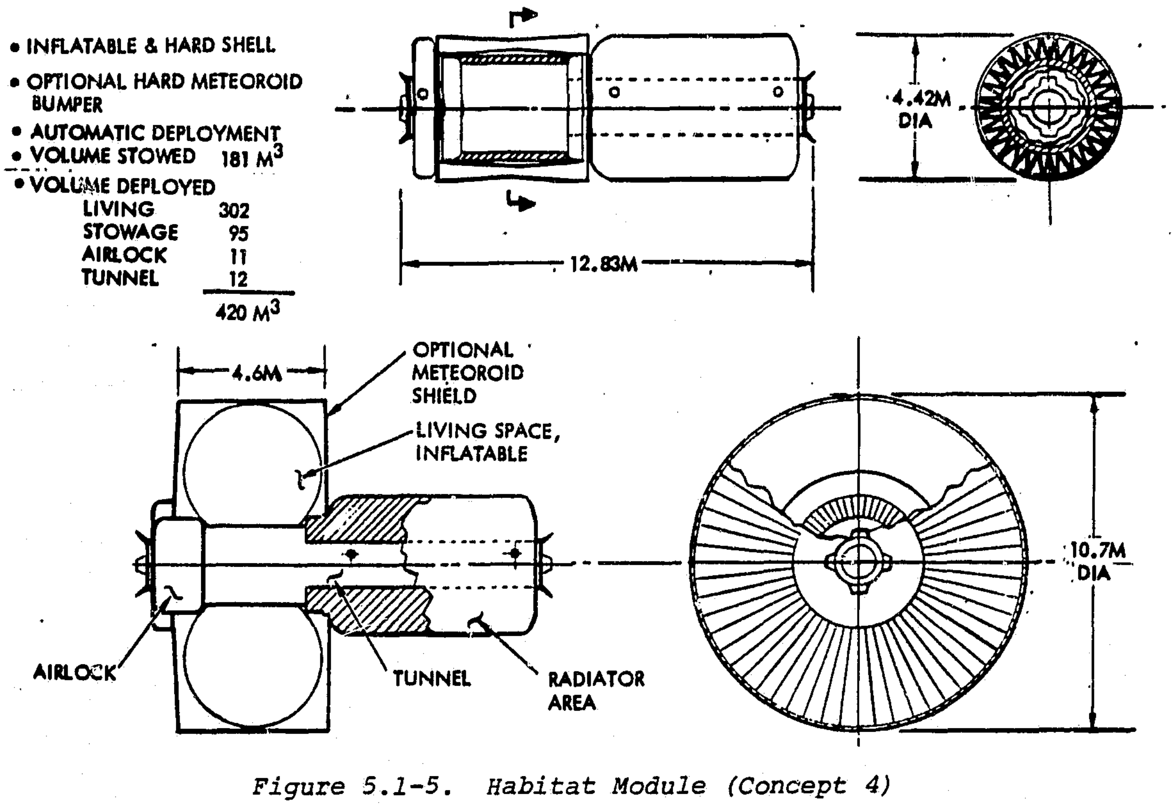

Figure 5.1-5. Habitat Module (Concept 4)

This module can be used with or without a hard meteoroid cover over the inflatable torus. The cover is folded and stowed around the outside of the collapsed torus. When the torus inflates, the cover unfolds and expands along with it. Deployment of the cover is completed when the panels which form its end walls are released and swing into position under the influence of springs.

The shape and size of the torus can be chosen to suit requirements. It can be much larger and it does not have to be a circular cross section. The shape, as drawn, is a large circular room (10.7 m diameter) with a column in the middle. The column is a portion of the tunnel which can be used to stow equipment for launch. The equipment can be subsequently moved into the torus living space.

5.1.5.3 Habitat Concept 5

When stowed, Concept 5 (Figure 5.1-6) resembles a standard habitat module (i.e., as currently designed for the Space Station) which is roughly the same size and shape as the orbiter payload bay. There are two sections which deploy from opposite sides of the module and form the living volumes. They are of excellent shape and can be subdivided to provide separate sleeping quarters, wardroom, working areas, etc. The whole module divides naturally into two spaces for safety/redundancy. At each end of the module is a large airlock and docking mechanism. Some of the space currently allocated for airlocks can be directed to stowage of items outside of the pressurized volume.

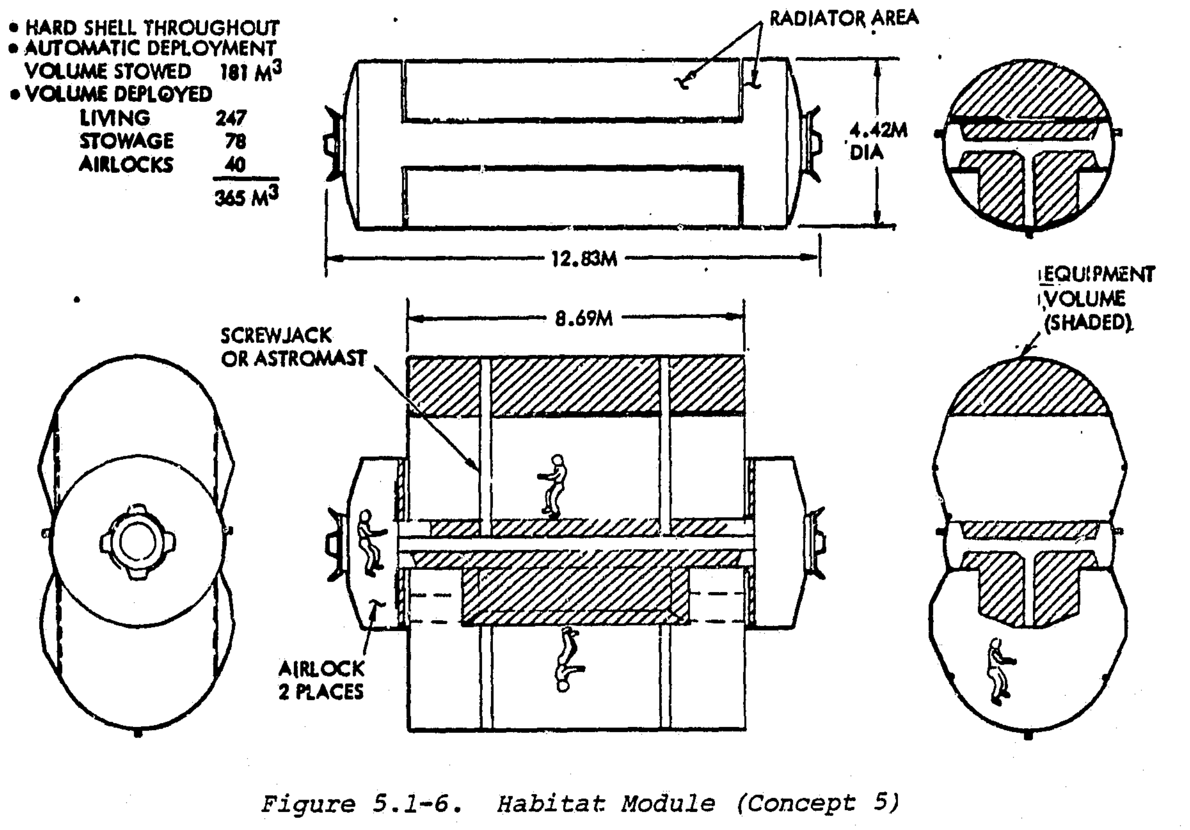

Figure 5.1-6. Habitat Module (Concept 5)

The structure is metallic, including the deployable sections and the close-out panels. The whole structure is supported in the orbiter by standard trunnions and a keel fitting. Expansion of the deployable sections can be obtained by either internal pressure or by mechanical means such as astromasts or screwjacks. Sealing between the deployable and the fixed sections can be realized either by a large internal pressure-tight bag and by sealing around individual joints. The area of the module available for use as a radiator is unusually large.

The ability to carry pressure-induced loads across the deployable joints is appreciated. These loads may be bending moments is addition to axial loads. EVA-installed, moment-carrying fittings are possibly needed with this concept.

5.2 TUNNELS

No study effort was devoted to tunnels since examination of Goodyear Aerospace Corporation accomplishments indicated ample work has been done relative to this component. The study time was better used in relation to hangars and habitats, for which no readily applicable effort is available.

5.3 OTV HANGAR

5.3.1 OTV Hangar Requirements

There are few really firm requirements for an OTV hangar beyond the obvious; i.e., it must remain in space for the same duration as the Space Station and it is launched using the orbiter.

Previous studies have indicated the infeasibility of pressurizing the hangar. The problems associated with opening/closing such a large pressurized volume are:

- o Sealing the hangar
- o The huge quantities of air lost each time the hangar is opened
- o Or alternatively, the very large power requirements if a pump is installed to recover the air before opening the hangar

The other hangar requirements listed are uncertain. They can be questioned:

- o Does the OTV require protection against meteoroids? Would it be more cost effective to accept a small risk of OTV damage than to build an expensive hangar?

- o If work platforms are required, is it reasonable to build a hangar for that purpose only; or would it be better to design the OTV so that astronauts attach themselves directly to its outside shell, rather than to hangar-mounted work platforms?

However, it was necessary to establish a basis for design studies, hence in spite of the lack of maturity in hangar and Space Station definition the following requirements were tentatively assumed:

- The OTVH size shall be suitable for the servicing/maintenance of an OTV. Assume an OTV size of 4.3 m diameter x 10 m long as a maximum, and work platform widths up to 2 meters.
- The OTVH will be unpressurized.
- The OTVH shall provide for a method of controlling the egress/ingress and stabilization of the OTV.
- Provisions shall be made for the following equipment:
  - Work platforms
  - Lighting
  - Electric power
  - OTV replacement items
  - OTV refueling
  - Minimum life support system (emergency)
- The OTVH shall provide radiation and micrometeoroid protection to the crew, OTV and equipment. (The end of the hangar pointing to earth is not subject to micrometeoroid bombardment and can be left open)
- Life in LEO for 10-20 years
- The OTVH shall be compatible with an orbiter launch

5.3.2 OTV Hangar Design Approach

There are two basic methods of designing the OTV hangar, i.e., inflatable or hard shell. The hard shell is a more conventional approach, and provides a firmer base for work platforms (reaction of astronaut loads). Its major problem is packaging in the orbiter and deploying from it. If the Hangar is to be packaged into a small portion of the oribter instead of using the whole of the payload bay, the problem of deploying it becomes even more formidable.

While an inflatable hangar is relatively easy to deploy, present designs may not provide adequate stiffness for the work platforms. There are various methods of hardening an inflatable structure such as foaming between walls, use of a skin material which hardens on exposure to radiation, and an erectile shell. In the erectile shell, a film of aluminum is added over the inflatable which is then pressurized to stress the aluminum past its yield stress. After the gas escapes the structure maintains its shape.

Regardless of which structural concept is used, all of the auxiliary structure and equipment are to be built into the assembly and deployed with it. These items are the crew work stations, lighting provisions, OTV ingress/egress provisions, OTV stabilization provisions, crew ingress/egress provisions, OTV refueling, and attachment to space station.

An alternative to automatic deployment of the hangar is erection or, more likely, a hybrid arrangement. The complications of automation must be balanced against the difficulties of EVA erection. If EVA erection is considered, it is better to use a method that requires a minimum of orbiter support.

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5.3.3 Meteoroid Impact Analysis

Since one of the purposes of a hangar is to provide protection for the OTV against the impact of micrometeoriods, it is pertinent to examine the potential damage to the OTV if an OTV hangar is not present.

The critical parts of the OTV are the LH_2 and LO_2 tanks which are contained inside an unpressurized structural shell as shown (Figure 5.3-1). Present designs (ground-based and space-based) utilize a graphite composite faced sandwich construction with a 0.035 gm/cm^3 (2.2 lb/ft^3) aluminum honeycomb core. A rigid polyurethane foam core is expected to be structurally adequate for a space-based OTV design and provides superior meteoroid shielding capability over that of the honeycomb core. The foam core does not have sufficient shear and compression stiffness to stabilize the face sheets against wrinkling during Shuttle launch of a ground-based OTV.

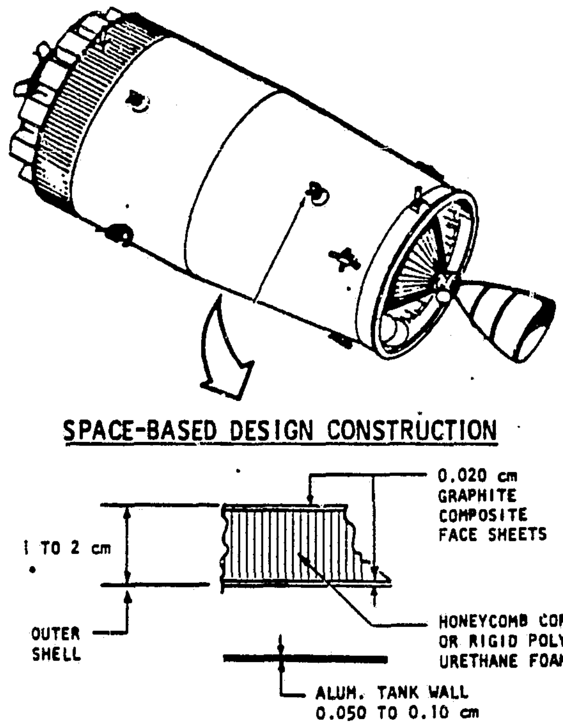

Figure 5.3-1. OTV Construction

The Rockwell Space Station studies are pursuing a space-based OTV which will be returned to earth after 16 missions (6 to 18 months). The micrometeoroid analysis for this design indicates there is a 1% probability the space-based OTV tank wall will be punctured in six months, and a 3% probability it will be punctured in an 18-month lifetime. This analysis is regarded as conservative since most of the tank wall is located a significant distance from the outer shell. Furthermore, even if a tank containing LH_2 or LO_2 is punctured by a micrometeoroid there will not be a catastrophic failure. The tank wall materials (2219-T87 or 2014-T6) are thin gauges and have a "leak before

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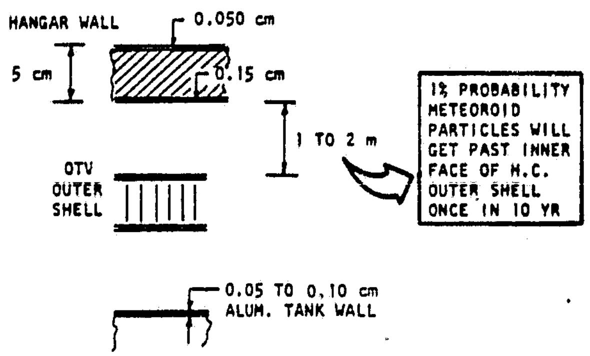
failure" characteristic that precludes explosive decompression. More than 100 cycles of pressure are required for a crack 0.04 cm deep by 0.38 cm wide to propagate to a through-crack.

The additional protection of a hangar wall for the OTV is illustrated in Figure 5.3-2. This figure also shows the cost implications of a hangar versus replacement of an OTV. Clearly, the cost of a conventional hangar cannot be justified on the basis of meteoroid protection alone. Only if the cost of launching a hangar can be drastically reduced by tight packaging in the orbiter or by other means, does the project become reasonable (if meteoroid protection is the only significant requirement).

ADDITION OF HANGAR STRUCTURE
(WITHOUT FOAM)



IS HANGAR REALLY COST EFFECTIVE IF METEOROID PROTECTION IS PRIMARY JUSTIFICATION?

ONLY WITH RELIABLE TIGHTLY PACKAGED CONFIGURATION REQUIRING 3 TO 5 m OF PAYLOAD BAY.

COSTS	($M)
OTV REPLACEMENT*	25
OTV LAUNCH	48
TOTAL	73
HANGAR LAUNCH	48
HANGAR COST	10
TOTAL	58

*SPACE-BASED DESIGN

Figure 5.3-2. OTV Meteoroid Protection/Hangar Cost

To date the studies of the Space Station have not produced hard requirements for an OTV hangar. However, recognizing the possibility that requirements may be developed, the OTV hangar concept development was continued.

5.3.4 Candidate Concepts for OTV Hangar

Seven OTV hangar concepts are presented (Figure 5.3-3). The concepts may be classified into three categories, i.e., deployable Hangars, rigid nondeployable hangars, and deployable/erectable hangars.

The four deployable hangar concepts, i.e., Concepts 1, 2, 3 and 7 are discussed below.

Concept 1 is a rigid shell which packages from a shape which is essentially cylindrical (8.2 m dia.) to a star shape which fits into the orbiter bay. An extendable truss which is used for ingress/egress of the OTV is part of the deployable shell. The stowed hangar occupies all of the orbiter payload bay.

Concept 2 is a rigid shell which consists of four half cylinders stowed in the form of one cylinder inside another. It, too, occupies all of the orbiter payload bay when stowed.

Concept 3 is an inflatable shell which expands from a module which contains the interface for docking to the Space Station, and the mechanism/structure for OTV ingress/egress. There are various methods which may be used to harden or rigidize an inflatable structure which otherwise might be too soft for a working interface. The stowed hangar occupies approximately 1/4 of the orbiter payload bay.

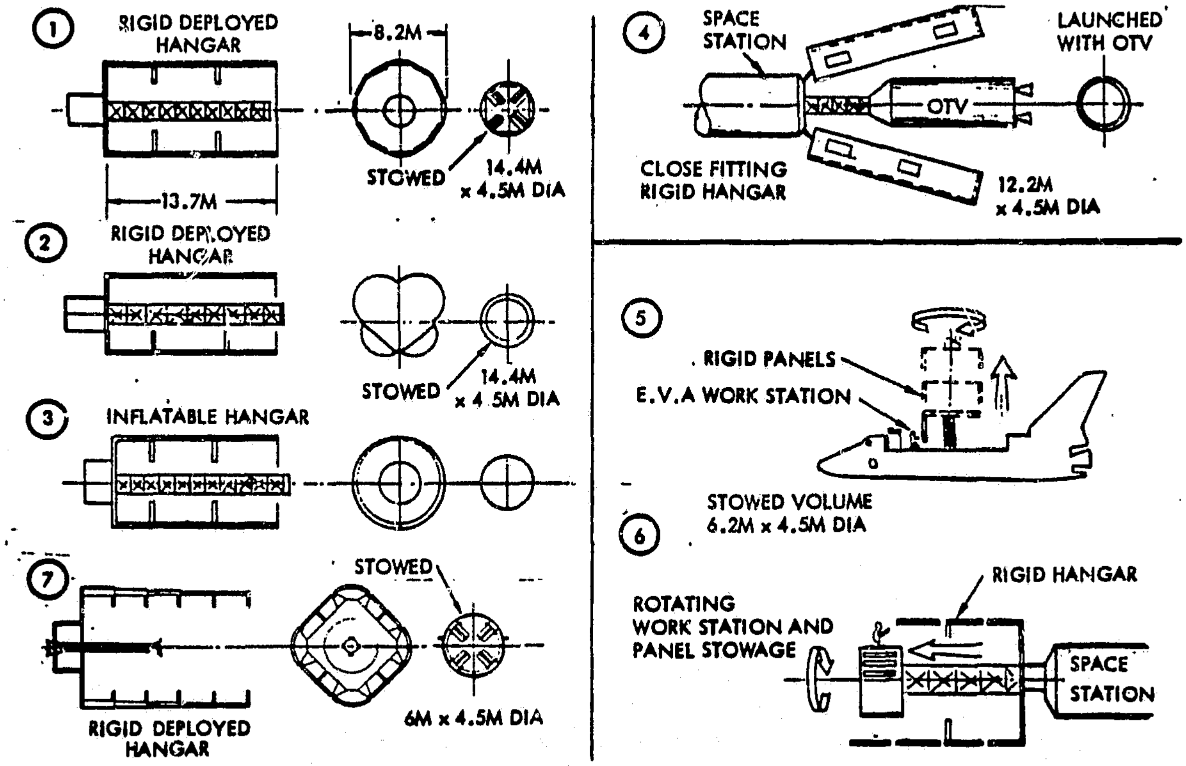


Figure 5.3-3. OTV Hangar Concepts

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Concept 7 is similar to Concept 1 except that it has a double fold (i.e., longitudinal as well as lateral) which enables it to occupy only a short length of the orbiter payload bay. (This concept is, subsequently, discussed in greater detail.)

Concept 4 is a rigid nondeployable hangar in the shape of a clam-shell which fits closely around the OTV. The OTV is launched in the orbiter inside the hangar, and exits from the hangar by means of an extending truss. Normal servicing/fueling of the OTV is performed through the aft end of the Space Station which is docked to the OTV. Access to other areas of the OTV is by doors and foot holds strategically located in the shell of the Hangar.

Concept 5 is a deployable/erectable system built out of the orbiter. A deployable section with an interface for the Space Station is raised incrementally by an extendable truss which also rotates about its major axis. There is an EVA astronaut work station in the orbiter bay with storage for a number of rigid panels which are attached manually to build up the hangar a it rotates and raises.

Concept 6 is similar to Concept 5 except that the hangar is constructed from the Space Station instead of from the orbiter. For both Concepts 5 and 6, it is estimated that approximately 1/3 of the orbiter bay is used for stowage.

None of these concepts represents design that Rockwell favors. In an additional study effort superior configurations can be developed. However, to provide further understanding of the design problems to be encountered, three of the foregoing concepts are discussed in further detail in the next section. (The drawings are provided in Volume II).

5.3.4.1 Hangar Concept 1

This concept (Figure 5.3-4) is a hard shell structure which folds laterally for stowage. When folded it occupies all of the orbiter payload bay. The operational sequence is:

- Use the RMS to remove the hangar from the orbiter
- Attach to the Space Station using the docking interface
- Unfold the panels of the main structure using the actuators built into the hangar
- The working platforms on the inside wall of the hangar may be deployed either by springs or by actuators

The extendable/retractable astromast is used for ingress/egress of the OTV, for which purpose it is equipped with a docking mechanism to interface with the OTV. Lighting power outlets and similar items may be built into the wall of the hangar.

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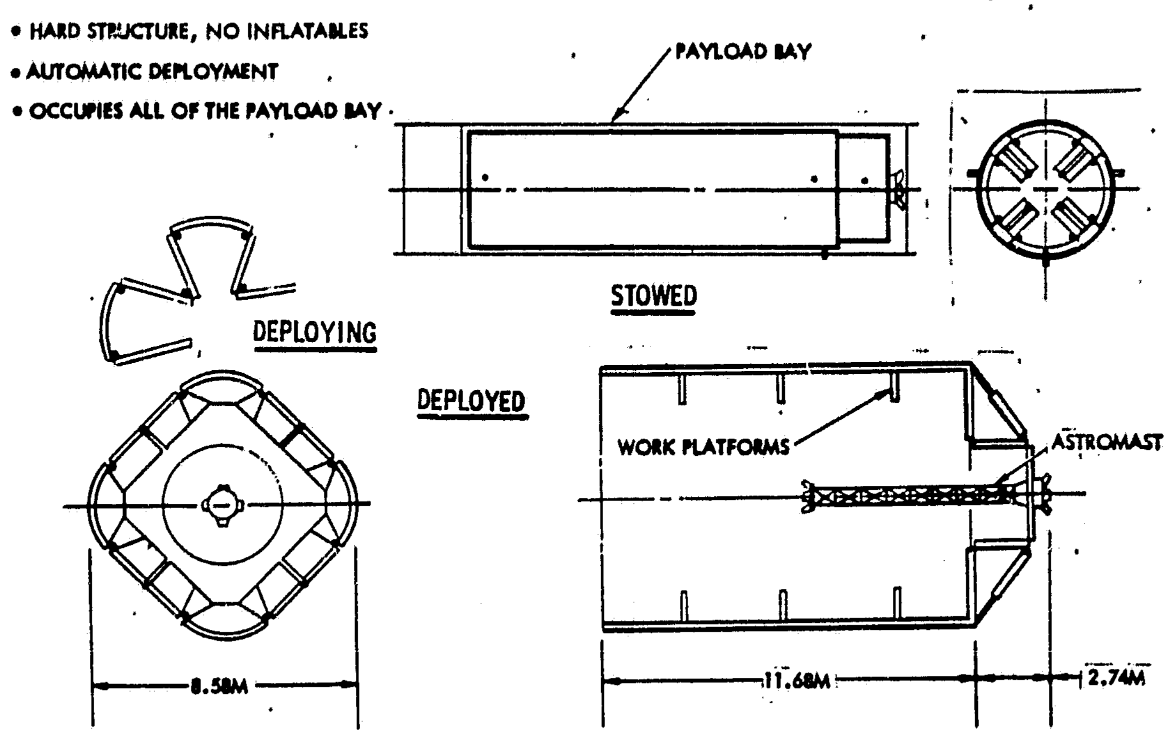

Figure 5.3-4. OTV Hangar (Concept 1)

Although the basic deployment is automatic, it is quite possible that EVA astronauts could profitably be used to lock the wall panels together for increased structural integrity, to install equipment, spares, etc.

5.3.4.2 Hangar Concept 6

This concept (Figure 5.3-5) uses EVA astronauts to erect the hangar walls around a deployable base. When stowed in the orbiter, approximately 6.0 meters of the payload bay is used. The stowed package consists of a deployable base or hub and cradle. The deployable base contains a docking interface for the Space Station, an astromast with a docking interface at its free end, deployable "umbrella type" ribs, and a hub structure. The cradle contains 40 separate panels which form the main structure of the hangar, a docking mechanism to interface with the astromast, a mechanism to dispense the panels as required during the hangar erection, and work stations/foot restraints, etc., suitable for EVA astronauts.

The deployment sequence is as follows:

- o Remove the two units from the orbiter using the RMS (the two units are coupled together by the astromast and temporary fasteners)
- o Attach the hub end of the hangar to the Space Station.
- o Automatically deploy the umbrella ribs.

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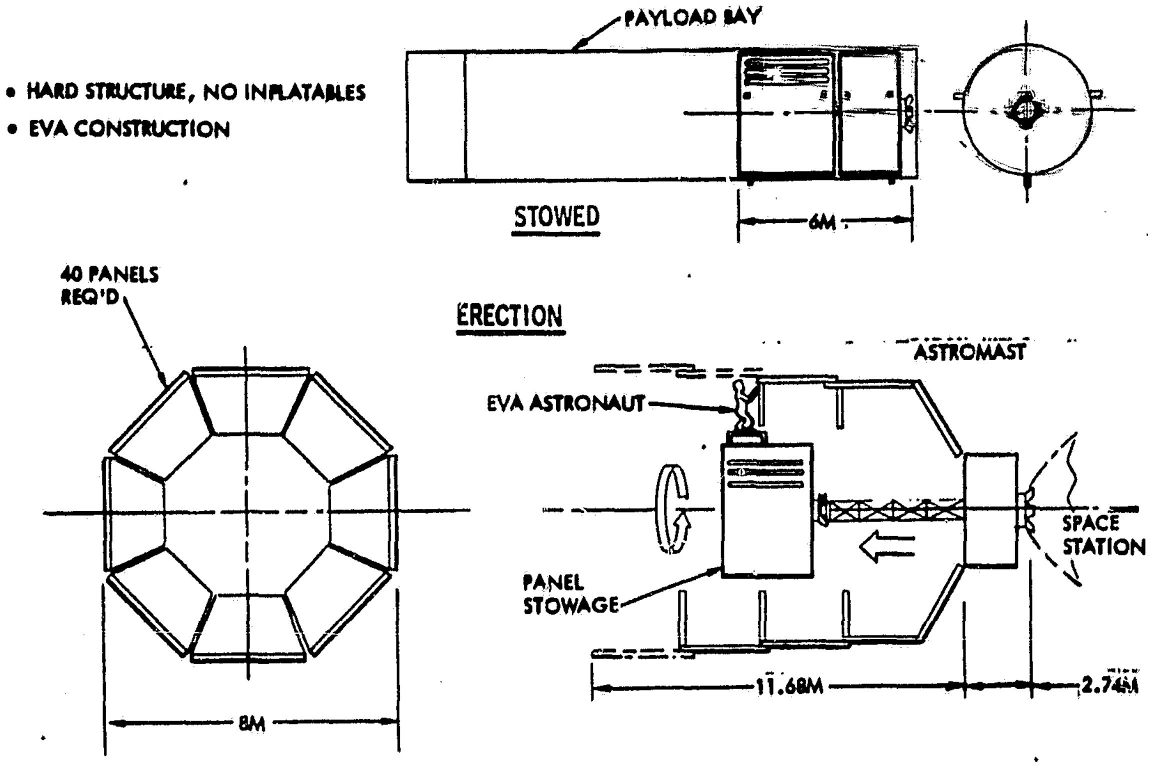

Figure 5.3-5. OTV Hangar (Concept 6)

- o EVA astronauts begin to remove panels from the cradle and form the hangar structure. The panels attach to the umbrella ribs to form the end wall of the hangar. The wall panels attach to the end wall and to each other.

- o The cradle rotates on the astromast and the astromast advances as required to enable the EVA astronauts to attach successive panels

- o The cradle is removed and returned to the orbiter (alternatively it may remain with the Space Station and be used for logistics stowage).

There are many methods which may be used for joining the panels together such as spring loaded "click-in" joints, hand-held power tools for tightening joints, over-center locks, or gang latches.

5.3.4.3 Hangar Concept 7

This design (Figure 5.3-6) is a development of Concept 1. The difference is the incorporation of a double-fold system (instead of a single fold) to reduce the space occupied in the orbiter payload bay. In this version, the stowed length is reduced from 14.4 meters to 5.6 meters with a significant cost saving. The installation procedure is:

- o Use the RMS to remove the stowed package from the orbiter
- o Attach the hangar to the space station using the docking interface

- o Deploy the first fold in the same manner as described for Concept 1
- o Deploy the second fold as shown in Figure 5.3-6

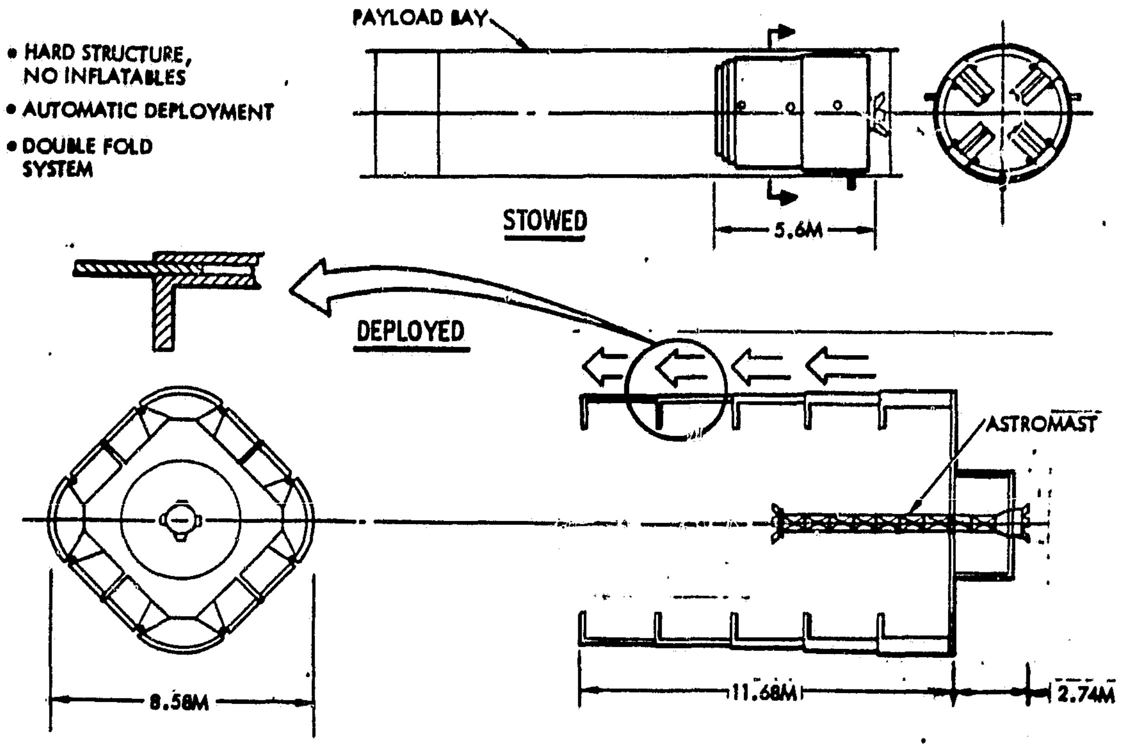

Figure 5.3-6. OTV Hangar (Concept 7)

The second fold is a series of telescoping sections of the main hangar wall. Extension of the telescoping sections can be accomplished by a series of stem actuators at each panel to provide a pushing action, or by the use of the astromast (with a suitable attachment) to draw out the telescoping sections.

It is recognized that the double folding of such a large volume poses formidable problems, and that much investigation remains to be done before such a system could be rated as reasonably feasible.

5.4 MAJOR DESIGN ISSUES

The major design issues that have surfaced during this study applicable to habitats and OTV hangars are listed below. The issue of OTV hangar justification is dependent on what probability of no puncture is desired, and the relative costs of OTV replacement versus the cost of a hangar.

Habitats

- o Means for installation of equipment in inflatable structures
- o Capability and frequency of repair of inflatable inner wall due to micrometeoroid puncture

- Rigidity of unpressurized inflatable structures
- Inflatable materials suitability in space environment for 10 to 20 years and compatibility with crew (non-flammability, non-toxic)
- Provision of radiation shielding for crew protection
- For hard metallic shells, the capability to sustain pressure-induced loads across deployable joints

OTV Hangars

- Requirements favoring the need for an OTV hangar have not been developed in Rockwell Space Station studies
- Justification for the hangar as a meteoroid bumper is dependent on the extent of design conservatism desired by NASA.
- Cost of OTV possible replacement vs. the cost of hangar development and launch
- Rigidity of inflatable OTV hangars relative to crew EVA imposed loads

5.5 POTENTIAL TECHNOLOGY DEVELOPMENT NEEDS

The potential new technology development needs that have surfaced during the course of this study are shown below. Rockwell is aware, through much of the design information gratuitously furnished by Goodyear Aerospace Corporation, that some of these items have been addressed by Goodyear to various stages of completion but to different requirements (in general, for smaller volume, reduced pressures, shorter lifetimes). The time devoted to this activity did not permit a screening of the applicability and status of these needs, but this should be done in future studies.

- Habitats Utilizing Inflatables
  - Assurance of no serious propagation of meteoroid puncture hole size
  - Repair of pressure containing wall
  - Micrometeoroid puncture resistance
  - Material suitability to crew and 10 to 20 years exposure in space environment
  - Maintainence of structural rigidity despite loss of pressure
- Habitats Using Metallic Structures
  - None applicable at this time

- o OTV Hangars
  - For inflatables—maintenance of structural rigidity compatible with crew-induced foot loads, equipment attachment, space platform inertia loads
  - On-orbit foam in-place techniques

5.6 CONCLUSIONS

The following major conclusions pertinent to the use of deployable volumes are evident from the foregoing discussions:

- o Application to habitats appears attractive—large useful volumes achievable.
- o Metallic structures (can be sealed) provide ample meteoroid radiation protection and equipment mounting surfaces, but are constrained by pressure loads/packaging requirements.
- o Metallic structures may require subsequent EVA for moment-carrying capability to withstand pressure.
- o Inflatables alone are not sufficient—hard structure required for mounting of consoles, orbiter and space station integration, heat rejection.
- o Inflatables in conjunction with rigid core module provide variety of feasible large volume designs, provided:
  - Materials are suitable to crew safety/space environment
  - Foam micrometeoroid stopping power is comparable to existing data
  - Repair of puncture or use of meteoroid bumper is feasible
  - Additional 0.24 $grms/cc^2$ for protection to crewman eyes is provided
- o Hangar requirements are ill-defined—OTV meteoroid protection alone is not sufficient justification.
- o Most attractive OTV hangar concept appears to be metallic deployable/erectable or inflatable with foam core (provided stiffness is adequate).

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